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United States
Naval Postgraduate School

INTERFACE OF MATERIALS AND STRUCTURES
ON AIRFRAMES

PART 3

DESIGN PROBLEMS IN AIRCRAFT STRUCTURES
INCLUDING
PROCEEDINGS OF MONTEREY SYMPOSIUM

Ulrich Haupt

October 1971

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NAVAL POSTGRADUATE SCHOOL
Monterey, California

Rear Admiral A. S. Goodfellow, Jr., USN   Milton U. Clauser
Superintendent   Provost

ABSTRACT:

The proceedings of the Monterey Symposium on Design Problems in
Aircraft Structures provide a basic survey of design problems
from the engineer's viewpoint. Further analysis of the present
situation draws attention to some essential aspects which are
not yet generally recognized. This leads to the conclusion that
recent design problems cannot be solved on a technological level
alone. An organizational effort is needed to disseminate available
information. Beyond this, the complexity of interactions must
be understood more thoroughly and this requires an educational
effort on a broad basis. A practical and systematic approach
toward the solution of these problems is developed.

TABLE OF CONTENTS

FOREWORD

This report forms the final part of the project Interface of Materials and Structures on Airframes and includes the Proceedings of the Monterey Symposium on Design Problems in Aircraft Sturctures.

The project Interface of Materials and Structures on Airframes has been sponsored by the Naval Air Systems Command under the cognizance of the Structures Administrator and has been conducted at the Naval Postgraduate School, Monterey, California. Two earlier reports were published as Part 1: Basic Design Considerations and as Part 2: Outline of Decision Process in Structural Design. They were concerned with basic aspects of the design process as they influence problems of interaction between materials and structures.

In order to coordinate these considerations with recent experience in industry and to recognize those design problems which most companies have in common, a symposium on Design Problems in Aircraft Structures was held at the Naval Postgraduate School in Monterey, California, on July 15 and 16, 1971. About twenty invited participants represented aerospace industry, government agencies, and research institutes. Prepared talks on particularly significant aspects of design problems were given by engineers thoroughly familiar with the present state of the art and discussion sessions followed.

The proceedings of the symposium are given in Section I and related questions with various comments are listed in Section II to stimulate general thoughts regarding design problems.

Section III contains some basic considerations and conclusions, attempting to bring together the viewpoints of design engineers, engineering management, and government agencies. It begins with a consideration of technological problems and recognizes that they are being approached systematically and competently on an engineering level. This leads, however, to the realization that such work on a purely technological level will not suffice to solve our design problems. Additional organizational and educational efforts will be required. Some of them have been recommended already by committees concerned with these problems. Some more aspects are added and an integrated approach is suggested on the final pages of this report (pages III-21 et seq.).

There is no consensus of opinions in this field. Conclusions must be based on the subjective interpretation of facts. They always have to be submitted to much discussion and careful examination. The considerations of Section III incorporate ideas developed by widely scattered people. Any shortcomings and

controversial aspects of the discussion are the obvious responsibility
of the coordinator of this project. The conclusions must not be
construed as necessarily representing the attitude of the Navy
Department.

The participants of the symposium and many individuals
throughout the aerospace industry made this report possible by
extending a spirit of full cooperation and giving generously of
their sparse time and hard-gained experience to discuss problems
which were not always easily defined.

SECTION I

PROCEEDINGS OF SYMPOSIUM ON
DESIGN PROBLEMS IN AIRCRAFT STRUCTURES

PROCEEDINGS OF SYMPOSIUM ON

DESIGN PROBLEMS IN AIRCRAFT STRUCTURES

The symposium on Design Problems in Aircraft Structures was held at the Naval Postgraduate School in Monterey, California, on July 15 and 16, 1971.

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WELCOMING REMARKS

M. U. Clauser
Provost
Naval Postgraduate School

It is with some feeling of nostalgia that I come back into this closer contact with men from the aircraft industry. After having spent quite a number of years building airplanes, I am glad to tell you how very pleased I am to welcome you here at the Postgraduate School and how appropriate I think it is to have a conference of this sort at this time. I am sorry that Admiral Goodfellow is in Washington and is not able to also extend his most hearty welcome to you. We are also very thankful to the Naval Air Systems Command for the sponsorship of this symposium and we welcome Commander Cundari in the efforts to organize and help sponsor this work.

I would like to take a few moments, because I have this sort of background and continuing interest, to speculate a little bit about the future but I would like to get a running start from the past. The 1940's were certainly the great days for the aircraft industry as I remember. There was the dictate of President Roosevelt that we were going to build 50,000 airplanes and it really happened during World War II. Then came the very interesting transition to jet airplanes and even rocket airplanes and I can remember the time I spent on the design of the Skyrocket airplane which held the world's altitude record and several speed records and how we had to work our way through some new thinking. Then, in the fifties, we turned to rockets with all the trials and tribulations and in the sixties we had the ability to really have ballistic missiles and space programs, with the emphasis on complex systems and ultra-reliability.

Now, as we come to the seventies, it is very interesting to see this return to emphasis on the aircraft but there are some trends which, it seems to me, pose a little bit of a dilemma. It is this problem of complexity. Costs have gone up as an exponential function. With our concern about the cold war and about our competition in the scientific war, as you might call it, and our space competition with Russia, somehow we always felt that a great nation could afford the increase in price. We seem to still want to make things even more complex but we must recognize that in the last few years the nation is no longer willing to support an exponential rise in costs.

If you add together the R & D costs on such things as DoD, AEC, and NASA, you get a nearly constant growth rate of about 20% per year from World War II to at least the middle sixties. But at the same time, the population was growing only 1 or 2%, and the gross national product only about 5%. This trend cannot continue indefinitely.

Further, we probably can no longer accept a reduction in the numbers of airplanes we buy in order to be able to obtain the greater complexity. The military services clearly cannot afford to procure just one very costly airplane. I think the new situation throws a double burden on the designer. He must now make the equipment more effective and at the same time cheaper.

There will be a great responsibility on the designer and many people will be watching as the realities of the situation do unfold. It is a real challenge to get people together who will think about how we are going to handle the complexity of design and at the same time can make the airplane continue to be more effective and not just keep going up in price.

Here at the Postgraduate School, as in a number of universities, we certainly realize the needs and problems which our graduates will face. We recognize the obligation to give them the intimate knowledge to cope with the decision-making and operating problems of the future. So it is a pleasure to find a way to interact with the aircraft industry more intimately and it is a pleasure to have you here. Anything we can do to make your stay here more profitable, more enjoyable, we stand ready to do it. I thank you very much for the opportunity to welcome you this morning.

INTRODUCTORY REMARKS:

THE NATURE OF DESIGN PROBLEMS

Ulrich Haupt
Naval Postgraduate School

Design problems in aircraft structures have existed as long
as there has been aircraft design. They usually found their
solution by following a general pattern along the lines of
engineering judgment and experience, experiments and analysis.
In case of special difficulties the specialist had to exert
special efforts, usually resulting in more experiments and more
refined analytical methods. Adding up all these efforts over
more than six decades, we see the results in form of jet trans-
portation and space flight and there is good justification for
taking pride in substantial accomplishments.

Now, however, we have reached a stage where recent experience
forces us to reappraise our situation. We feel that we are close
to the boundaries of our present concepts and that the complexity
of our problems is somewhat frightening. In the past, we could
approach a problem by isolating it and applying all the available
expert knowledge and experience. This is still the case for many
of those present and future problems which are the bread and
butter of the engineering profession. Yet the really challenging
problems which we have to face become increasingly difficult and
usually we cannot isolate them anymore. They are actually of a
different nature.

Think of our situation in fatigue. We began by isolating
the problem of crack initiation. This took us to the problem of
rate of crack propagation, then to the problem of fail-safe
design, then to the problem of residual strength and fracture
mechanics. In each one of these fields we have to deal with a
subset of problems -- including material selection, structural
concepts, stress level and stress concentration, manufacturing
methods, load spectrum, and service life. Fatigue is a problem
of detail design but it frequently has its roots in early
decisions far upstream in the design process and may result in
catastrophic consequences due to a special combination of
circumstances way downstream in the service life. All of them
are interwoven.

In addition to such problems of detail design, we have
another kind of problems which are typical of advanced design.
Examples are:

- How do we define the criteria when we want to apply
  newly developed materials to present-day aircraft --
  considering weight, cost, risk, life cycle, uncertainties
  of load spectrum, etc?

- How do we optimize materials and structural configuration
  for thermal and chemical environments as complex as we
  have to face them for supersonic cruise or space shuttle?

These challenging problems in detail design and in advanced
design have in common that they incorporate a large number of
parameters. Most of them are interdependent and many of them are
not clearly defined at all. Beyond these aspects of interdependence
and quantity, a new qualitative aspect is introduced. Up to now,
the main emphasis in aircraft has been on airworthiness. For
our new type of design problems, however, we have to combine
airworthiness with design optimization under very complex
conditions. This takes us into a new field. For both detail
design as well as advanced design we have to develop an outlook
which is much wider than the conventional knowledge, experience,
and expertise of the specialist. This question of basic outlook
is closely connected with design problems.

*   *   *

It may be helpful if we look at developments which have
become visible in engineering education. Throughout the 1950's
increasing emphasis was given to science curricula until we
realized in the later 1960's that engineering aspects had been
seriously shortchanged. We came to understand the basic
difference between science and engineering. The scientific
approach starts from a given problem and proceeds by analytical
and experimental methods -- simplifying and clarifying a problem
to its skeleton in order to establish and understand basic
principles. On the other hand, the engineering approach envisions
a goal and proceeds by defining the problem, creating alternative
possibilities, analyzing them by scientific methods and making a
decision about the optimum solution -- taking into account all
the inherent complexities and practical consequences in order
to find a practical answer.

So we finally began to realize that engineering is more than
applied science. The engineer must have a different attitude and
a different viewpoint than the scientist. Slowly, much too
slowly, there is a growing emphasis on creative engineering,
value judgment, and interdisciplinary responsibilities. It
will take some time until this process becomes clear and the
results will penetrate into industry, but a new trend is visible.

*   *   *

There seems to be an analogy between the situations in education and in industry. Science and engineering in education have their counterparts in the professional activities of analysis and design in industry. In education we are realizing that engineering incorporates science as a most essential ingredient but has to reach into wider aspects. Correspondingly in industry design may include analysis as a most essential ingredient but again it has to reach into wider aspects. However, I should not belabor this point because some of my friends hold different opinions on it.

There is full agreement that our analysts have been at the forefront of developments during the last few decades. In the field of structures they have, among many other things, developed a beautiful system of finite elements for basic airworthiness calculations and they are working on mathematical aspects for structural optimization. Yet our actual difficulties are not covered by these somewhat abstract considerations of airworthiness and optimization. Our difficulties are of a very practical nature with an endless number of real-life complexities. Beyond analytical aspects we have to

· recognize any possible mode of failure;

· evaluate uncertainties and risks;

· establish a value system for any optimization procedure.

Here we have design problems in the fullest and widest sense of the word. For their solution we require an analytical mind coupled with an imaginative spirit. This is expressed in the designer's intuition as it has been applied in the past when the designer could handle a few parameters on his sliderule. Yet for our present design problems with an ever-increasing number of parameters we have to develop a new methodology.

Such a new methodology is a challenge to do in a systematic way what we have been doing intuitively. Two different aspects can be considered:

· firstly, a technological effort to reduce the risk of structural failure;

· secondly, a methodical effort to establish a process of decision making under complex conditions.

Regarding the first aspect, namely the technological effort, a decisive step is taken in the new Air Force program on Advanced Metallic Structures. This program is directed toward improved technologies in materials, structures, and manufacturing and

emphasizes the importance of a systematic transfer of information. It is an eminently practical, hardware-oriented program. As such it does not sponsor any ideas which are still somewhat vague in themselves.

Regarding the second aspect, the methodical effort toward the decision making process, this is unfortunately still in a category of vagueness. Perhaps it depends mostly on becoming aware of the problem and on communicating about it. There will not be any quick and ready solution but it seems that fundamental elements could crystallize sooner if a common concern would be developed. A great effort is exerted by specialists regarding mathematical aspects of optimization but before we can ever hope to use any mathematical refinements, we have to clarify some of the most basic aspects of the decision-making process.

Decision making is an essential part of design. Many variations can be found in other fields and it should be easy to point out how primitive the state of the art is. From the highest level of deciding about national policies and war or peace to the very personal level of choosing a marriage partner, important decisions are being made in a somewhat haphazard way. Any self-respecting gambler would estimate his chances more thoroughly.

There is nothing mysterious about decision making and optimization. The basic requirements are an adequate problem statement and objectivity -- and this is where we usually fall short. Clearly defined values as well as uncertainties have to be included. The values will generally be of a variety of dimensions -- weight, cost, time, reputation, etc. Utility theory or similar methods of decision making provide a basic tool to express these multi-dimensional values on a single scale. The corresponding detail manipulations can be left to operations analysts but every designer will have to acquaint himself with the underlying concepts.

Notions like uncertainties and value systems still have a long way to go before they become household words in aircraft design. Yet it is the designer who has to establish a merit function, basing it on his experience and making it clearly visible so that any parameters can easily be submitted to scrutiny and modification. This field is still wide open and largely unexplored. It concerns both detail design and advanced design and represents design problems of a very different nature from those to which we have been accustomed.

In the subsequent presentations and discussions we want to stay at first on familiar grounds and consider some specific problems of technology which are of much concern for new materials,

including fatigue, fracture mechanics and fail-safe. Afterwards
we will see whether we can proceed toward some basic aspects of
methodology. This should take us to some ill-defined problems
which, nevertheless, may be of fundamental importance.

There are many viewpoints which sometimes look quite
different but frequently can be reconciled when some good will
exists and a real effort is made. Good will can be created as
soon as we become aware how important a problem is and how much
we need a solution. How successful our efforts will be -- that
is often beyond our jurisdiction. All we can do is to make a
sincere attempt -- and that is the purpose of this symposium.

TRANSLATING HIGH-STRENGTH STEEL CHARACTERISTICS
INTO EFFECTIVE STRUCTURAL DESIGNS

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ABSTRACT: Development of test methods to characterize high-
strength steels is discussed up to the present fracture
mechanics methods. These test methods are related
to service experience by specification requirements,
structural criteria, and design practices. Examples
are shown for parts which failed unexpectedly--pointing
out the need for materials and structures engineers
to improve their ability to communicate with design
engineers.

I would like to review briefly some of the attempts of the
materials engineers to characterize high-strength steels for the
design engineer. For the purpose of this discussion, high-strength
steels will be defined as those having an ultimate strength of
200,000 psi or above, used in the range of -65° to 400°F. These
steels have yield-to-ultimate ratios from .80 to .98 with minimum
elongations of approximately 5% in 2 inches.

My basic thesis is that we have not provided the designer
with many quantitative numbers to work with for load-carrying
capacity designs. We give him tensile strength and modulus and
that is about all he can put into a formula. When you talk about
things like ductility, impact strength, corrosion resistance,
these are just comparative values. There is no ready way to put
them into a numerical calculation and use them.

I would like to go back, since I have about as much gray
hair as most of you here, and think of the 30's when we looked
at the mechanical properties, including impact and ductility,
of a new steel. If they were similar to one which had already
a lot of good service life, you assumed that you could endorse
it for design. Then, when we got into the war, much acceleration
went on, and we used a big variety of steels with a lot of odd
combinations of chemistry because of alloy shortages. We worked
pretty much from the basis of tensile strength and hardenability,
and we used all kinds of things which probably would make us
shudder today. We also worked with a lot of specialized tests,
including notched tensile fatigue tests. But again, we had a
hard time relating those numbers to a specific design and
idealized specimens bore very little resemblance to actual parts.

Now the latest thing is the fracture mechanics approach and here we are getting a little closer to the numerical idea. But the first thing for which we have to characterize a steel is K_{ic} and many of you know the war which is going on about the type of specimen to use and the wide variety of values for a given heat of steel. The designer is pretty much at the mercy of an empirical system when he is making a part out of a slightly different cross-section or form. He cannot make a clear-cut decision without a lot of specialized help and specialized work in the laboratory, and even with all that, we still have failures.

In the early 1950's, we got enamored with the idea that we could take some of our more conventional steels, specifically good old SAE 4340, and move them up to the 260,000 psi heat-treat level. A classical paper by Melcon of Lockheed-California* was published in 1953 and many of the aircraft companies, including mine, did work of their own and decided that this was a promising approach to get the weight down. We proceeded on the F-8 Crusader airplane, which was designed in the 1953 period and flew in the early part of 1955 and which had a considerable amount of steel parts in the 260,000 heat-treat level, particularly in the landing and arresting gear systems.

We had some problems and I would like to review several typical examples to show you how the designer and the materials and structures engineers can get themselves into a trap. I am sure all of you have dozens more in your own experience. Some of the problems occurred fairly early in the programs; others took time to develop in the environment in the fleet.

Fig. A-1 shows the main landing gear inner cylinder shell, made of 260,000 heat-treat steel, with a clevis at the upper end. The interesting thing in this design was that the designers found the space in this upper cylinder could be used as a pneumatic bottle for an emergency actuation of flaps and gears in case of a hydraulic failure. This was not part of the landing gear system at all. Every time the engine was running, the cylinder was pressurized to 2000 psi and when the engine was shut down, it bled off slowly by leakage. Explosive failure occurred not during a landing but while the airplane was being pivoted on the carrier deck around one wheel which resulted in maximum stress at the clevis.

We did not have an electron microscope at the time but it was obviously a very brittle failure. All the standard tests had been performed but we never had a failure nor any problem at

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*Melcon, M.A., Ultra High Strength Steel for Aircraft Structures, Product Engineering, October 1953.

all with this part in the plant at any time. No matter what we did, we could not explain at that time exactly what was going on. Being 260,000 heat treat, we did not use electrolytic cad plate on it. We knew about hydrogen embrittlement, so we tried to keep away from that area. At that time, vacuum cadmium plating was not developed and so we just had an organic finish on these parts.

Subsequently, when we got the electron microscope several years afterward, we found out that some minor scratches in the paint had permitted the part to corrode in a very small area, about 10 or 15 thousandths in diameter. The little bit of corrosion resulted in enough hydrogen to hydrogen-embrittle the part in that local area and also to give it some stress corrosion. The designer had veen quite ingenious but the combination of maximum stress in the clevis, hoop tension due to internal pressure, H-embrittlement, and minor stress corrosion resulted in failure. The part is now vacuum cadmium plated and painted and is no longer pressurized as a pneumatic bottle.

Fig. A-2 shows the horizontal stabilizer shaft which is heat-treated to 260,000. Through a series of offset tapered pins, a little over an inch in diameter, it is attached to the actuating horn of Fig. A-3. We found a few of these shafts cracked and after some checking -- it was also in the pre-electron microscope era -- we decided that a washer at the tapered pin might be getting misaligned. So we designed a specially shaped washer to fit the inside contour of the shaft and to avoid local high stresses. We also added some more paint in local areas and did not seem to have any more problems.

Then, about six or eight months later, we found some cracks again. We could not put our finger on the definite thing but we made a few more changes and still have had occasionally some cracked parts. Examination of these later parts by electron microscope revealed hydrogen embrittlement and we have tried to trace the source of it and to keep control over the processing. The part is vacuum cadmium plated and does not get any electrolytic cadmium. So here we have a design which, from the designer's point of view, looked like an efficient and good way to get high strength and close tolerance and where impact and ductility values did not indicate any reason why this would not work.

Fig. A-3 shows the stabilizer horn which is also a 260,000 heat-treat part. For this part it turned out that the 260 high heat treat was specified because the structures engineers were told by value engineering that this would cost hardly any more. Like most structures people, they were looking to a growth version of the airplane and left an extra margin where it can be done without overpenalizing design and weight. So this

is one part where we moved the heat treat back to 200. With
4340 steel you have to stay out of the 220-240 range for
metallurgical reasons, just as a precautionary move.

Then, recently, we had a failure again. The reason was
typical for the way things can happen. We had a new vendor
who came out a little high in hardness. He pulled the test
part which went with it, got the ductility, and felt there was
no problem because he was giving a little more than required.
However, when we dropped the heat-treat to 200,000, we decided
that we could electrolytically cad plate it because, generally
speaking, there is not any hydrogen embrittlement problem with
200,000 heat treat. Now with 246,000 heat treatment and electro-
lytic cad plating, we had hydrogen embrittlement coming out the
ears. Again the designer did his job but there was not enough
communication about the processing precautions with the
manufacturing people.

Fig. A-4 is a swinging arm for the arresting gear at the
260,000 heat treat level. It gets a certain amount of impact
and we had failures as shown. This was a good old traditional
case where we had a big discussion among the structures, design
and material engineers about how much interference fit these
pressed-in bushings could have without getting above the thresh-
hold stresses for stress corrosion cracking. However, in cad
plating there was some tolerance which could get involved
occasionally. You will say in pressing them in you should have
shaved the excess cad plating off. But we had a manufacturing
operation which deep-freezed the bushing and dropped it in.
So they shrunk it down to undersize and did not have to get the
cad plate shaved off but when it came back to room temperature,
we were above the stress allowables by just enough to get us
into trouble.

This part is now 200,000 heat treat. There seems to be
some magic about this number. Things like hydrogen embrittlement
and stress corrosion are not very critical at this level. For
any bushings which we have now in high-strength parts, both
aluminum and steel, we try to prevent water entrance. We provide
a section around the bushing for a sealant groove to keep the
water out of the interface between the bearing and the fitting
even though it is press fit. We forbid deep freezing bushings
because of the tolerance and condensed water problem.

Fig. A-5 is a 260 H.T. bellcrank on a droop mechanism. It
was designed for stiffness and did not need the 260,000 strength
but it failed in a fatigue sense. This was pretty much an
interior part when it was on the deck and the droop was retracted.
However, there was full flowing air going around this part
during the extended position and it got enough corrosion to

provide a pit. Again, the heat treat was dropped down to 200,000 and then the corrosion effects did not appear to be as critical in fatigue. We also went to more protection, did a little aluminum spraying in the beginning, and then went finally to electrolytic cad.

Fig. A-6 seems to be hard to explain. This is the main fin beam in both the F-8 and A-7 airplanes and it is about 40 x 35". It is forged from a 4340 billet and in the center where normally the worst material would be, the part is split in two directions. There are holes drilled into it, and it has only primer for protection. The part is 260,000 heat treat, has 3 million flight hours on it, and has never failed. By the way, the note "crack" in the figure only refers to a specimen which they tried to straighten during a heat treat operation.

Now you could say that the reason for no failure is that this part is lowly stressed or is in a mild environment. Well, this part does not have a sustained load like in a pressure cylinder or in a pressed-in bushing. It is designed for a gust condition which rarely occurs and the total spectrum life is easy. But since you have to stay out of the 220-240 range for this type of steel, the next jump down to 200 would have been too big. The design also has no sharp corners and the holes are straight for standard close tolerance bolts. Besides, it is an interior part in warm, dry air and is protected from the engine heat by a stainless steel shield. Even with all these good things I am still surprised that we have a zero failure rate.

This last example shows that you should not panic if a catastrophic failure occurs on some part at the same strength in the, same steel. We had some other parts in the airplane which were not loaded any higher but were in more exposed areas and failed. It is hard to tell the designer when he is really in trouble about corrosion unless the environment can be very accurately defined.

I might say that we have had some cleanliness problems in steel but we tried to control them by specifying values for reduction of area in specified locations in each heat of steel. This seemed to give the fewest problems along the way in unpredicted failures. We felt this was our protection but, at the same time, how do you explain an empirically picked number to the designer?

Well, I just wanted to show you some histories of how we have given the designer some problems and have complicated them right along. Designers and structures engineers need to understand the effect of stress raisers, residual stress due to

processing and induced assembly stresses due to clevis type
parts or other misalignments. Effects from stress corrosion
and hydrogen embrittlement have to be considered in selecting
strength level. When working with high-strength steels,
process engineers, designers, tool planners, and manufacturing
personnel need to be educated in the catastrophic effects of
grinding, plating, cold-straightening, cleaning, weld repairing,
etc., without proper post stress relief treatments.

The designer also needs more education along the lines of
NDT so that he knows what defect size can be detected. Even
if we are great believers in magnaflux, you still can magnaflux
a part and not find a crack which is there. I think, almost
intuitively, our own people have backed up from welding ideas
in high strength steels quite a bit. Talking about steels of
300,000 psi, they are wary of sustained stresses and are trying
to keep the stress level down at a fairly conservative level.
That hurts from the weight and cost point of view pretty fast.
We worry not only about things that happen in our own shop
but we are also indoctrinated by the troubles everyone else
has. Risks are involved in this area and when you want to move
off to something new, you have to convince not just your engineering
and manufacturing management but you are involved pretty strongly
with the operating management even when they are not engineering
oriented.

Most failures in high-strength steels result from three
major sources, namely: (1) improper design; (2) improper
processing; (3) undetected flaws. Now we are giving the designer
a new input from fracture mechanics, with new tools and new
concepts and there is no table where he can look up the answers.
Few failures are the result of fracture toughness per se but
its effect on premature failure must be understood.

I would like to conclude in saying that there is no
substitute for communication. We have to get engineering,
manufacturing, and quality control together but I think we
also will have to develop a more definitive and quantitative
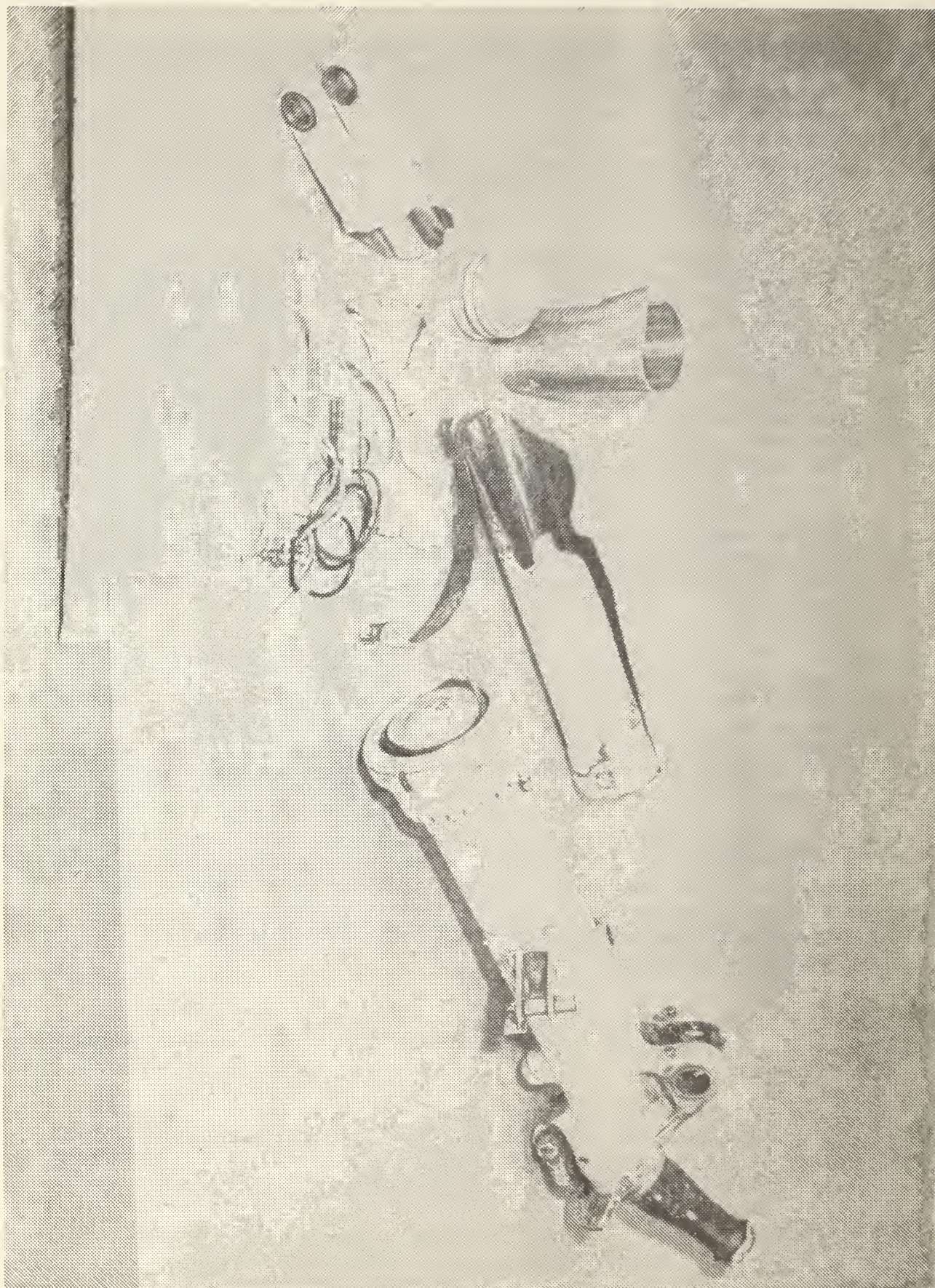way of characterizing high-strength steels than the present
empirical data.

Figure 1 – Inner Cylinder, Main Landing
Gear Inner Cylinder

Figure 2 - Unit Horizontal Stabilizer Shaft
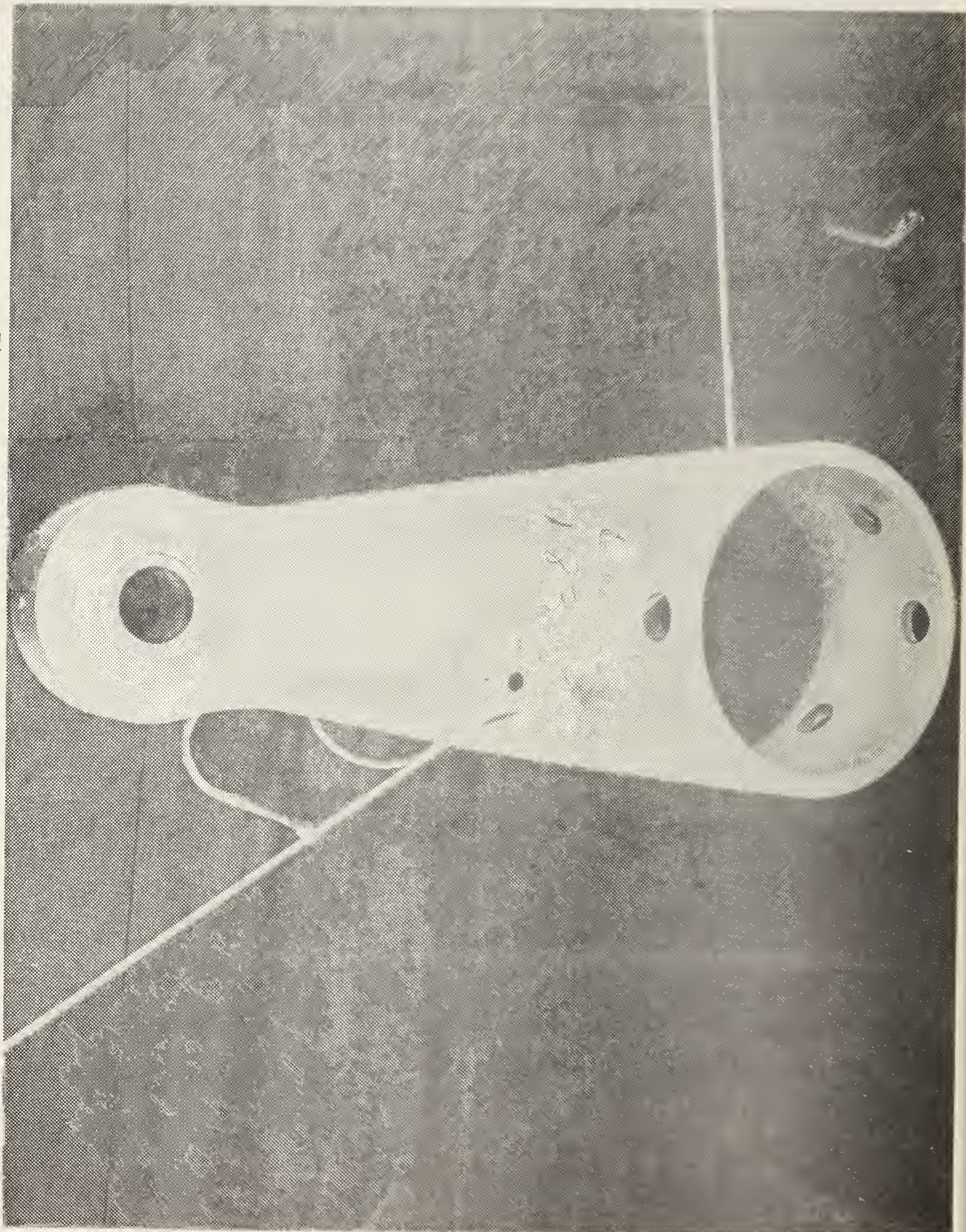
Figure 3 - Unit Horizontal Stabilizer Horn

Figure 4 - Arresting Gear Swinging Arm

Figure 5 - Droop Bellcrank
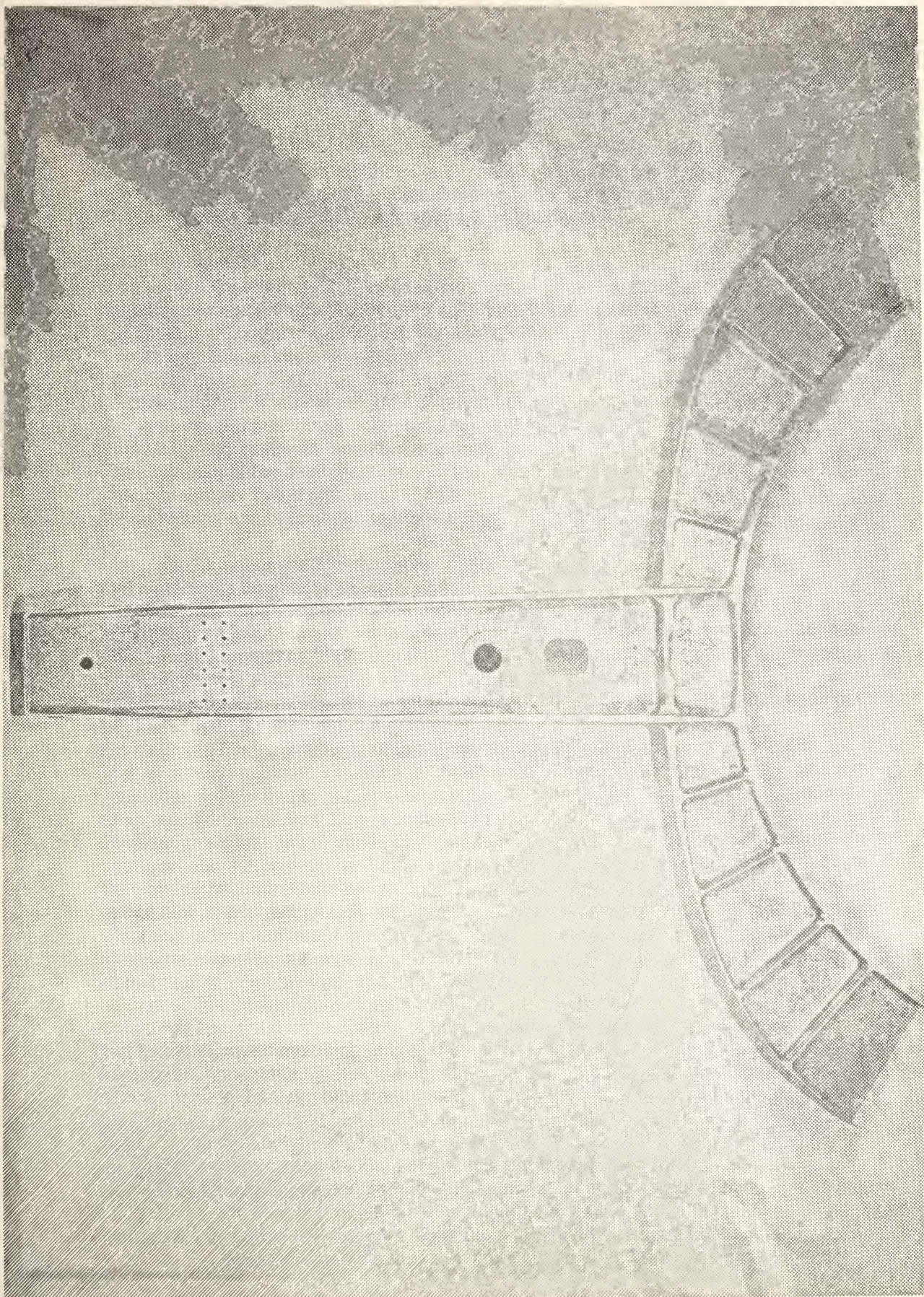
Figure 6 Beam, Vertical Stabilizer

FATIGUE -- RELATING PAST EXPERIENCE
TO DESIGN

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Abstract: Three general approaches are used to design aircraft structures for a long service life: control of design details, establishment of design stress levels, and analysis combined with experiments. These approaches are discussed and illustrated by examples from previous experience. Examples include the use of the stress severity concept and how it relates to fatigue life, fatigue quality index, and design stress levels of similar structures -- emphasizing that determination of safe life is not yet an exact science and must be based on previous experience.

The previous speaker gave us some examples of how to make use of experience. As you know, there is no good theoretical approach for predicting fatigue life of an aircraft component and so, in order to circumvent this problem, previous experience is used to form a baseline from which to work.

Structures which have performed adequately over a long period of time provide assurance that a similar design for a similar application will perform just as well. However, problems arise when trying to exploit new materials and new technology where previous experience is lacking. In this case, experience must be supplemented by adequate test data, although it is not always easy to obtain test data applicable to what is going to happen ten or twenty years hence. In this presentation I would like to discuss how our previous experience is used to the best advantage in coming up with new designs and how it relates to the overall problem of design, detail design and advanced design.

Figure B-1 illustrates how service experience is utilized in the design of new aircraft and where the information is used in relation to each phase of the program. A satisfactory service life can be achieved by approaching the problem on three different levels:

a. Control of aircraft design details -- based on service experience of design details from previous structures;

b. Establishment of design stress levels -- based on stress levels utilized in previous aircraft structures;

c. Theoretical and experimental considerations -- for new designs where previous experience is not available.

For any new aircraft structure each of the three above approaches must be utilized to achieve a good design from a fatigue point of view. Therefore, I would like to illustrate the use of these three approaches with some examples.

In the first approach, information gathered from experience has to be presented in a simple format and brought to the designer's attention so that he will look at the data and try to avoid these problems. Figure B-2 shows a common type of design detail in connection with access holes. Small holes in the vicinity of a bigger hole inevitably mean superposition of stress concentration. An acceptable solution can be achieved for many applications by eliminating the fastener hole in the vicinity of highest stress concentration due to the cut-out. For the preferred solution of using a clamp-on door, consideration must be given, of course, to the effect of fretting fatigue due to rubbing action between the clamp-on door and the structure.

Figure B-3 shows another type of problem which is often encountered in service and which consists of cracks in the corners of door cut-outs. Attempts have been made to try to reinforce these areas to reduce the stress levels and prevent the crack; however, this has not always been successful. The best approach in this case is to make a fairly generous radius in the corners of these types of cut-outs, using the rule-of-thumb shown in Fig. B-3. With this type of design concept, fewer cracking problems have developed in service.

Figure B-4 illustrates another area which may easily be overlooked as not a structural problem. A forging of this type needs to be attached to some type of jig for machining to final dimensions. Tooling holes are drilled at various locations for the purpose of holding the forging to the jig during machining. Often these tooling holes are located where the bending stresses are essentially zero. However, the shear stresses in a beam are usually high where the bending stresses are low and fatigue cracks due to high shear stresses have occurred at these locations. Also, often the tooling holes do not have as good a surface finish as attachment holes in other parts of the structure. For example, tooling holes may have scratches or other imperfections which amplify the stress concentration effect of the hole. As shown in Fig. B-4, plugging tooling holes with interference fit fasteners can increase the fatigue life 4-fold while reducing the fatigue quality level, K_{Test}, to less than 4.0. On our new designs we prefer to reduce the stresses by reinforcing locally around the hole.

Figure B-5 refers to a problem which is often overlooked. This is fatigue cracking as a result of deflections from one part of the structure induced on another part of the structure. In this example the nacelle fillet region had a doubler attached to the nacelle skin to prevent localized buckling. To prevent fatigue cracking at the end of the doubler it was necessary to extend the doubler and install additional rows of fasteners to stiffen up the area.

These simple types of examples provide the designer with some basic information as a first line of defense against fatigue cracking problems. Next, let us consider the selection of design stress levels. There are two ways of specifying design stress levels, either by putting limitations on the design ultimate tensile stress of the material or on the stress due to 1g loading conditions.

Figure B-6 shows one technique of selecting preliminary design stress levels. Based on service experience on previous aircraft, it relates maximum allowable design stress level to the number of flights when fatigue cracking problems were encountered. Most designs fall within the cross hatched band, therefore the percentage of ultimate tensile strength which can be utilized for a new design depends on the fatigue life that is desired in future service.

The results of spectrum fatigue tests of components can also be utilized to aid in the selection of design stress levels. The fatigue quality index, K, is determined from fatigue test results as illustrated in Fig. B-7. The analysis is based on the test spectrum which was applied to a given structural component up to the time when a fatigue crack was initiated, and a set of S-N curves for various K_t values representative of the material in which the crack was started.

With these data, a fatigue analysis is conducted for several K_t values, determining Σ n/N for each case, and finally calculating by interpolation on the K_t value which corresponds to Σ n/N = 1. This K_t value is designated the fatigue quality index K, of the structural component.

A number of components have been analyzed and compiled as shown in Fig. B-8. This Figure shows the distribution of K values obtained from 42 test results. The fatigue quality index ranges from slightly below 3.0 to above 5.0, with a mean of 3.65 or 3.7. So we can say that the fatigue quality of aircraft structures is something a little worse than an open hole which would have K_t = 3.0. For a new design, you might not want to use the average K value as a basis for structural design since you would have only a 50% chance of passing a fatigue test. Therefore, we usually try to design for a fatigue quality level

of about 4.0. In some areas of the structure a poorer quality
or higher K value can be tolerated because the stresses have
to be low for reasons other than fatigue. This type of analysis
is just a one-parameter (K) analysis. There are a lot of other
variables that have an effect on fatigue life, but with 43
results, you really do not have enough data to consider more
variables in the preliminary stages of design.

Figure B-9 shows how to use the K value to arrive at a
preliminary design stress level. For a new application, you
first develop spectrum loading conditions which the structure
is likely to encounter during its service life. Then you
perform a fatigue analysis with the same S-N data that was
used for the analysis of fatigue test results. If various
materials are being considered, you can conduct the analysis
using S-N data for the various materials to obtain a relationship
between the design stress or reference stress and the fatigue
quality index as shown in Fig. B-9. For this particular example,
if you picked a K value of 4 as being the type of quality you
think can be achieved in the structure, then a 43,000 PSI
stress level would be permitted for 2024-T3 aluminum alloy.
This analysis was conducted for a particular service life so
that the values in Fig. B-9 are all for the same number of
flights or the same number of flight hours anticipated in
service.

Assuming you have done the best job you can and are coming
up with satisfactory design stress levels and design qualities,
however, there are still new materials, new fasteners and new
approaches that you want to apply to the structure for which
you have no previous experience. In this case, you will have
to conduct various types of tests to develop information and
correlate the data with similar tests on structures in which
you have confidence. So, let's look at some of the types of
tests that are conducted and the results of some example cases.

Figure B-10 shows some results of two-row lap joint tests
with aluminum countersunk fasteners. The peculiar S-N curve
is the result of three types of failure. In the low-cycle
high-stress region, the failure occurs where you would normally
expect it, i.e., originating at the center of a fastener hole
where the highest stress concentration is located. In the
transition section, failure occurred in the countersunk sheet
away from the fastener holes, and in the high-cycle low-stress
region, the failure took place in the plain sheet also away
from the fastener holes. In the high-cycle region (> 300,000
cycles), the cracks initiated at the edge of the attachment
holes in the interface between the two sheets. This is where
the greatest rubbing action takes place and failures were caused
by fretting fatigue. The data in Fig. B-10 illustrates dangers
of extrapolating from previous experience, say less than

300,000 cycles, into a different region of life expectancy. For this example, you would get over a 50% reduction in stress level over what you'd expect with a normal extrapolation from low cycle part of the S-N curve.

Fatigue tests have been conducted on simple lap joints using lubricants, shims, etc. to try to eliminate the fretting fatigue type failure. Aluminum bronze between mating surfaces worked pretty well, teflon films did not work very well. However, for each application, tests have to be conducted to find out each technique that will work because with fretting fatigue the same technique doesn't always work for different applications.

Figure B-11 is another example of a fretting fatigue failure. This is the case of a countersunk hole in the skin of the fuselage and the example shown is from a pressurized panel test. Fatigue cracks initiated in the black area in the transition region from the countersunk to the cylindrical portion of the hole opening due to the flexure and fretting action when the fastener was rubbing against the sheet material in this general region. To avoid this type of problem, it is necessary to install more fasteners or larger fasteners to reduce the bearing and bending stress.

Figure B-12 shows the results of a series of tests on three-row lap joints conducted to investigate what type of fastener might be best for a given application. None of these failures represent bolt failures. The normal theory for the design of mechanically fastened joints is that if you use interference fit fasteners, you are going to get an improvement and if you use high clampup, you are going to get an improvement. So, combining these effects, you should get a terrific improvement. Oddly enough, the results did not come out that way. We got a very small amount of difference between the various fasteners tested. Only the two upper fasteners gave a little better result; however, it still isn't a dramatic improvement. The main reason for this anomally, I think, is again that fretting fatigue failure is coming into the picture and it is limiting the degree of improvement you can get by changing the fastener and the interference. Therefore, the state of the art of the material fastener system was pushed to the point where fretting fatigue type failures were occurring and something else must be done to avoid this problem.

Figure B-13 shows the results of some other tests, where a series of tests were conducted to evaluate various design alternatives in order to come up with a good design. This example is a typical fuselage joint encountered in an aircraft structure. A continuous stringer is attached to the skin and the continuous frame inside of the stringer is attached to the

skin by an angle which is cut out in the stringer region. The first concept was the simplest one without any attachment between stringer and frame. Crack initiation was at 10,000 cycles in the angle at the rivet attachment to the stringer.

The next design used a slightly different stringer shape which allowed a rivet attachment between stringer and frame. This gave an almost 9-fold improvement in crack initiation time. The crack was in the stringer at the rivet attachment to the frame which moved the fatigue critical point away from the skin.

A third design resulted in an additional 10-fold improvement in crack initiation time. This was due to an added clip on the side of the frame so that the load was more evenly distributed across the stringer. This shows how simple joints can be developed in the laboratory by running simple tests. This is probably the best way to go in many cases since the analysis is unreliable and rather difficult. It does indicate what good detail design can accomplish for you.

Another approach for coming up with a good design is to use finite element analysis techniques. Figure B-14 shows four different designs for a wing-to-fuselage joint with tapered stringers. The first two designs have a combination shear and tension connection, the last two designs are double shear joints. Normally, a tension-shear type connection is not a very good design from a fatigue point of view, but you can make it work provided you put enough material in the right places. For this particular application, it was considered because of the simplification of final assembly.

The finite element analysis takes into account the flexibility of the fastener systems, so the load transfer from skin to stringers is more or less properly accounted for. The resulting stress concentration or stress severity factors are shown at various places in Fig. B-14. The finite element analysis model is very useful for looking at various methods of shaping the material and for reducing the stress concentration down to a minimum. It permits one to come up with more or less an optimum design for each concept instead of running many fatigue tests. The example shows how a good design can be evolved using computer analysis techniques in conjunction with a minimum amount of test data.

The finite element type of analysis results in a stress severity factor and does not really predict the lifetime. So a correlation was developed between the stress severity factor (obtained from finite element analysis) and the fatigue quality index which is obtained by fatigue analysis of test failures. The correlation is shown in Fig. B-15 which indicates that a stress severity factor of about 3 corresponds to a fatigue quality index of about 4.

After design details and stress levels have been established and basic testing has been conducted, the final proof comes by conducting a full-scale fatigue test on the complete aircraft structure. There you integrate the effects of adjoining structure which you cannot simulate adequately in the laboratory on components. Fatigue is not an exact science yet, but much can be learned from experience and applied to new types of design. The use of this experience in conjunction with a suitable fatigue test program will provide a reasonable assurance of adequate service life.

INFORMATION USED	METHOD OF ACHIEVING REQUIRED SERVICE LIFE	PHASE OF PROGRAM

OLDER STRUCTURE GOOD FATIGUE QUALITIES PROVEN IN SERVICE → DESIGN NEW STRUCTURES EQUAL OR SUPERIOR TO OLDER STRUCTURE

PRELIMINARY DESIGN

ADOPT ALLOWABLE STRESS LEVELS FROM PAST EXPERIENCE → PROVIDE SUFFICIENT MATERIAL TO KEEP STRESSES BELOW ALLOWABLES

CORRELATION BETWEEN TESTS ON PREVIOUS AIRCRAFT STRUCTURES & SERVICE EXPERIENCE → CONDUCT FATIGUE ANALYSIS AND TESTS TO DEMONSTRATE THAT SERVICE BEHAVIOR WILL BE SATISFACTORY

DESIGN DEVELOPMENT AND QUALIFICATION

SERVICE EXPERIENCE

OPERATION

HOW SERVICE EXPERIENCE IS UTILIZED
IN THE DESIGN OF NEW AIRCRAFT

Fig. B-1

AVOID PLACING CUTOUT ATTACHMENTS AT POINTS OF
HIGH STRESS CONCENTRATION

Fig. B-2

AVOID SMALL CORNER RADII AT SHELL CUTOUTS

PASSENGER BAGGAGE AND CARGO DOOR CUTOUTS

AVOID

R – 3/8" to 1 1/4"

LOADING DIRECTION

ACCEPTABLE

IF R/H + R/W > 0.6

R 4" MIN

W   H
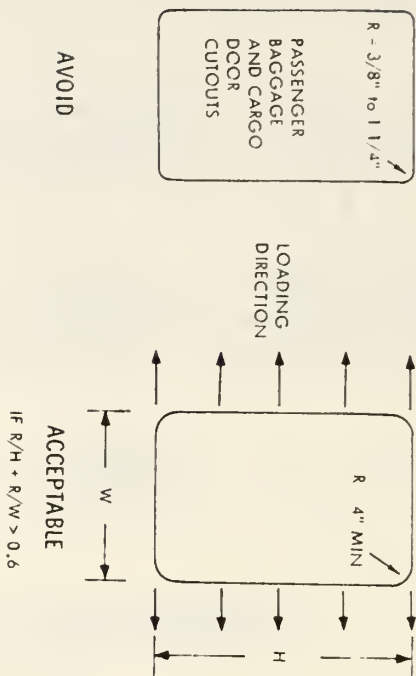
Passenger, baggage, and cargo door cutouts with corner radii in the range of 3/8 to 1 1/4 inch have in the past produced considerable fatigue cracks at the corners. Patches and steel doublers at the corners did not stop cracking. Sharp corner radii must therefore be avoided.

MAKE R/H - R/W > 0.6 WHENEVER POSSIBLE.

When R/H - R/W is smaller than 0.6, the frame around the cutout must be designed so as to keep the stress concentrations to a minimum.

The corner radii should be proportioned to the size of the cutout and the material should be distributed so as to avoid eccentricities and discontinuities. The working stress should be kept as low as possible at the corners.

Fig. B-3

TOOLING HOLES IN MACHINED FRAME FORGING

TOOLING HOLES PLACED NEAR THE NEUTRAL AXIS OF AN I-BEAM CAUSED FATIGUE FAILURE DUE TO THE SHEAR STRESSES IN THE WEB

ORIGINAL DESIGN
OPEN TOOLING HOLE.
K TEST 4.7
LIFE N

REPAIR SCHEME
TOOLING HOLE REAMED AND FITTED WITH INTERFERENCE FIT PLUG.
K TEST < 4
LIFE > 4N

PREFERRED DESIGN

WING CENTER SECTION BEAM

TOOLING HOLES

Fig. B-4

NACELLE

FATIGUE CRACK

WING-NACELLE
FILLET

A

A

WING CONTOUR

NACELLE
SKIN

FILLET ATTACHMENT SCREWS

NARROW FILLET SKIN DOUBLER
(2 ROWS OF SPOTWELDS)

FILLER

FILLET SKIN

NACELLE
DEFLECTIONS

WIDE FILLET SKIN DOUBLER EXTENDED TO HERE
(4 ROWS OF SPOTWELDS)

SECTION A-A

LOCAL BUCKLING OF THE WING-NACELLE FILLET, INDUCED BY DEFLECTIONS OF
THE NACELLE, CAUSED FATIGUE CRACKING IN THE FILLET SKIN AT THE EDGE OF
THE NARROW FILLET SKIN DOUBLER (2 ROWS OF SPOTWELDS). THE DESIGN WAS
IMPROVED BY THE USE OF A WIDE FILLET SKIN DOUBLER (4 ROWS OF SPOTWELDS).
NO CRACKS OCCURRED WHERE THE WIDE FILLET SKIN DOUBLER WAS INCORPORATED.

EFFECT OF INDUCED DEFLECTIONS

Fig. B-5

Fig. B-6

I-32
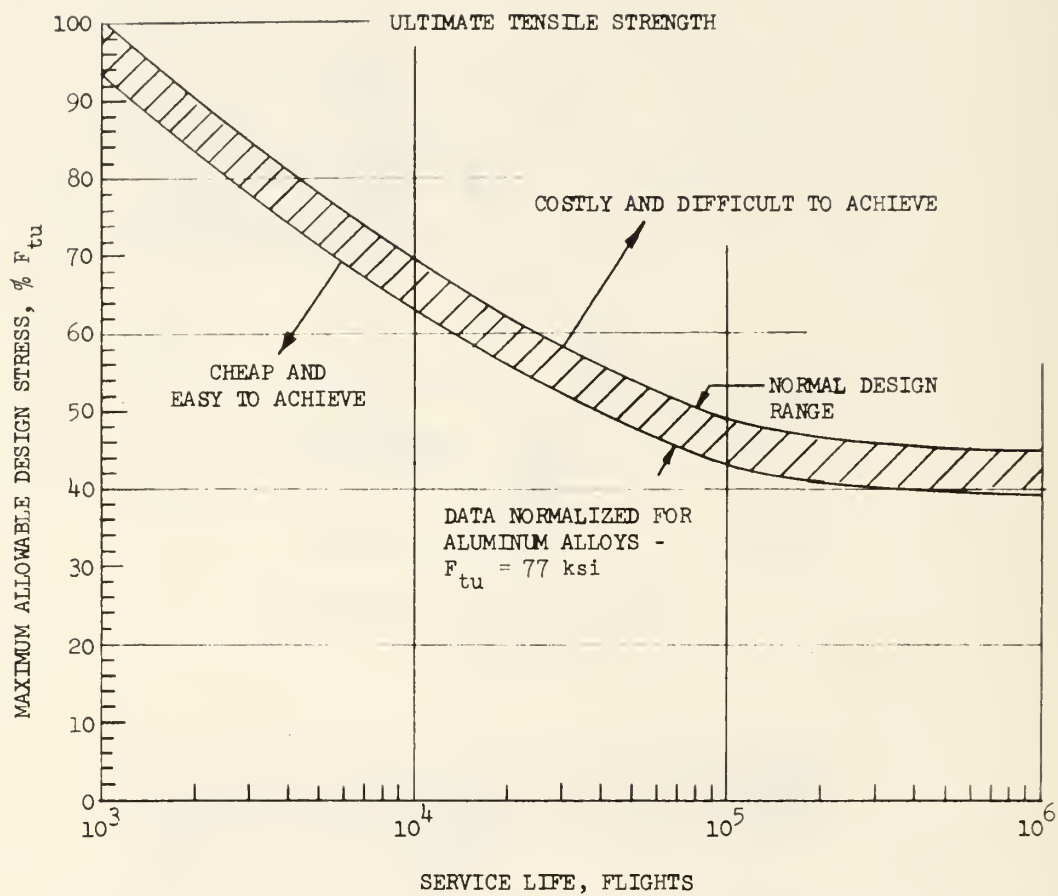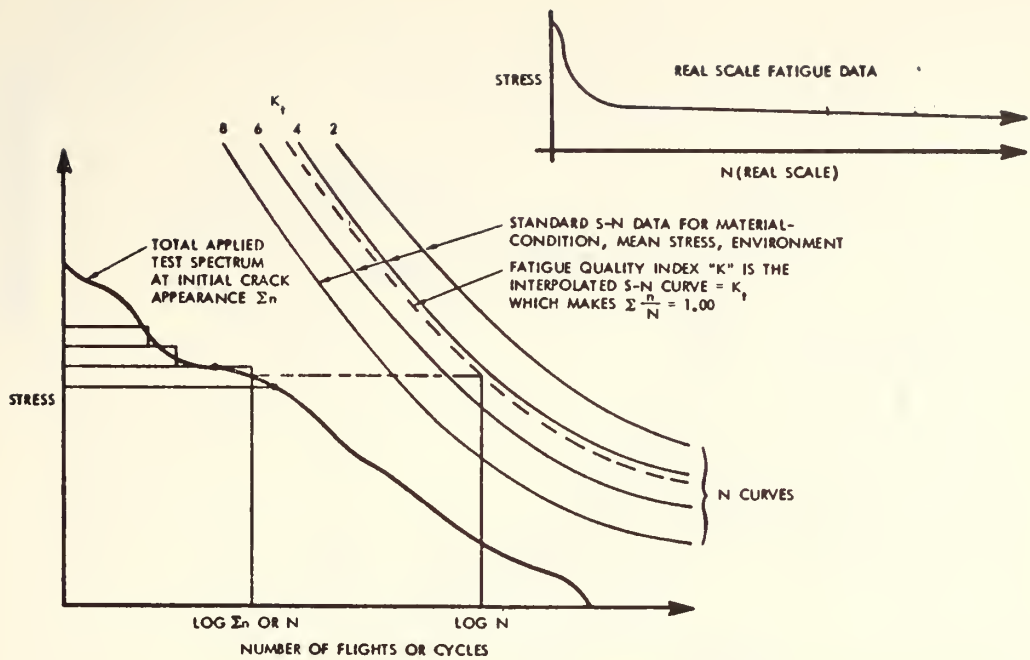
LABORATORY TEST DETERMINATION OF FATIGUE QUALITY INDEX

Fig. B-7
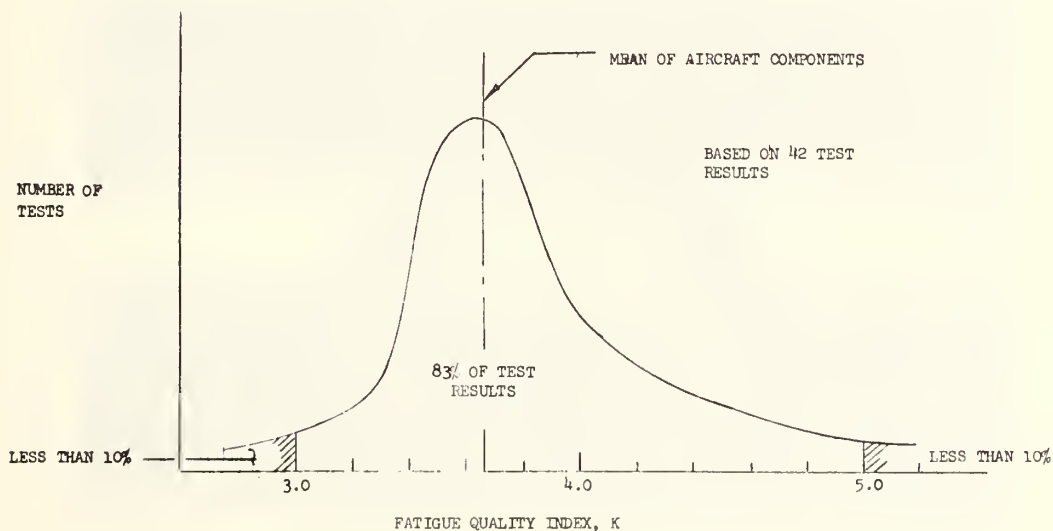
DISTRIBUTION CURVE FOR AIRCRAFT STRUCTURAL COMPONENT TEST VALUES OF K

Fig. B-8

FATIGUE CRITICAL LOCATION ON TRANSPORT
WING LOWER SURFACE

PERMISSIBLE
DESIGN
AREA
FOR
2024-T3

2024-T3

7075-T6

7178-T76

DESIGN ALLOWABLE TENSION STRESS, KSI

FATIGUE QUALITY INDEX~K
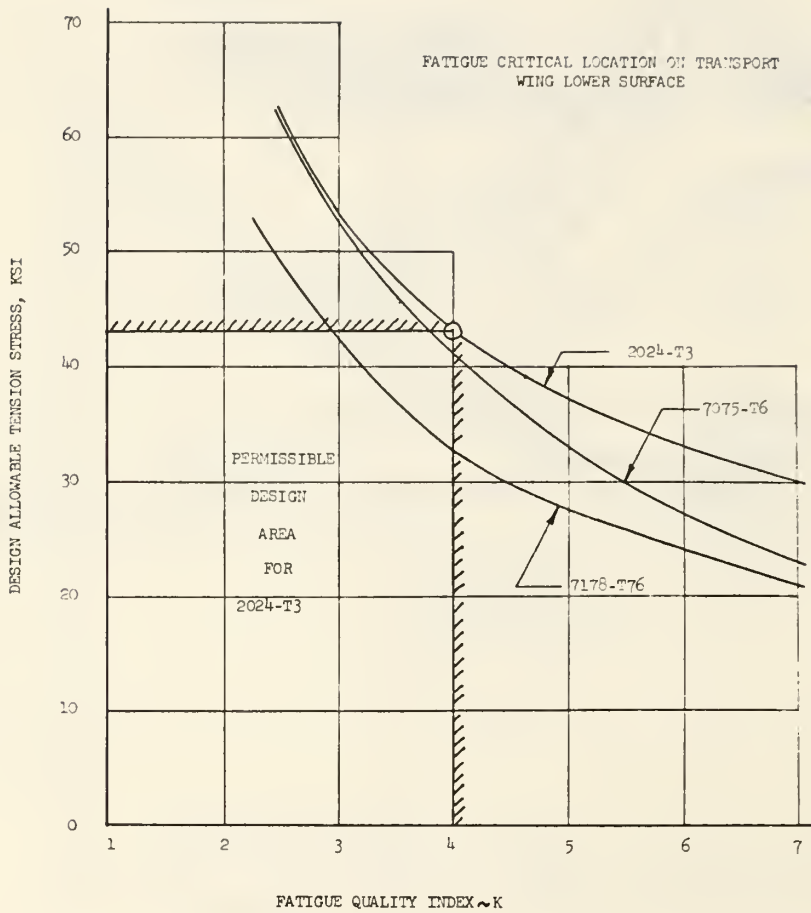
RELATION BETWEEN FATIGUE QUALITY INDEX-K
FOR A GIVEN FATIGUE LIFE

Fig. B-9

FAILURE MODES FOR LAP JOINT SPECIMENS
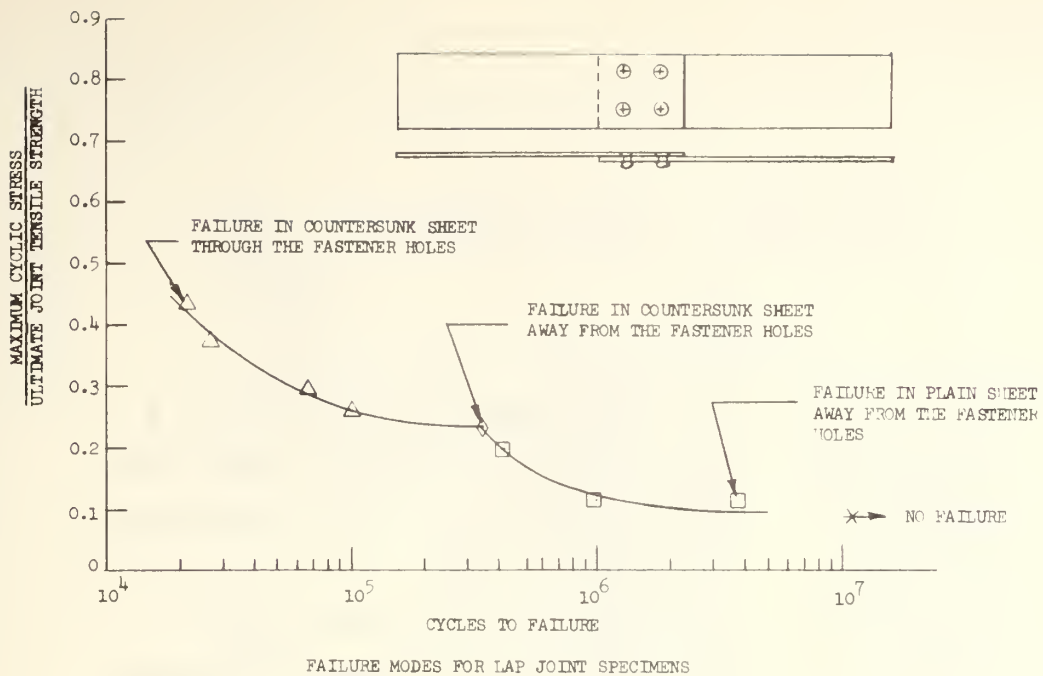
Fig. B-10

FRETTING FATIGUE

Fig. B-11

Ti-1-8-5 TLB-960	80,120 (Standard Interference)	
HL-51 Hi-Lok	52,680	(Clearance Fit)
Ti-6-4, TLV-100 (Taper-Lok)	44,510	(Std. Interference)
Broach Bolt	42,465	
Ti 6-4, HL-11V (Hi-Lok)	41,370	(Clearance Fit)
TL-100 Taper-Lok	38,330	(Standard Interference)
Huck Crimp	36,580	(Ream and Deburred Hole)
Steel Hi-Tigue	34,920	(Interference Fit)
HL-51 Hi-Lok	34,870	(Interference Fit)
Ti-Hi-Tigue	33,410	(Deburred Holes - Interference Fit)

0   50   100

FLIGHTS TO FAILURE - 100

FLIGHT-BY-FLIGHT SPECTRA FATIGUE TEST RESULTS OF
TITANIUM AND STEEL FASTENERS
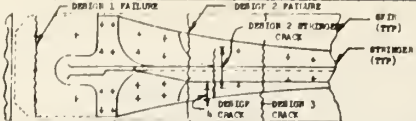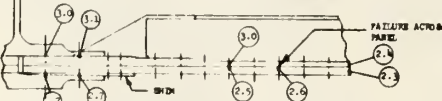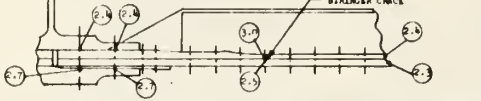
Fig. B-12

TYPE OF DETAIL	SPECIMEN CONFIGURATION (CONSTANT AMPLITUDE LOADING AT R = 0)	CYCLES TO CRACK INITIATION	CYCLES TO FAILURE
NO ATTACHMENT BETWEEN FRAME AND J-STRINGER STRINGER ANGLE FRAME END VIEW	CRACK IN ANGLE AT RIVET ATTACHMENT TOP VIEW ANGLE — STRINGER FRAME SIDE VIEW	10,000	63,800
Z-STRINGER RIVETED TO FRAME STRINGER RIVET ATTACHMENT END VIEW	TOP VIEW CRACK THRU STRINGER AT RIVET ATTACH- MENT TO FRAME SIDE VIEW	88,700	119,000
* CLIP ANGLE END VIEW	(TOP VIEW SAME AS ABOVE) CRACK INITIATION IN STRINGER AT RIVET ATTACH- MENT CLIP ANGLE FAILURE THRU STRINGER AT RIVET ATTACHMENT TO FRAME SIDE VIEW	888,000	998,000

* STRINGER AND CLIP RIVETED TO FRAME

DEVELOPMENT FATIGUE TESTS OF FUSELAGE DESIGN DETAIL

Fig. B-13

TYPE OF JOINT	SPECIMEN CONFIGURATION	TEST LIFE FLIGHTS TO FAILURE*
TOP VIEW OF DESIGNS		
ORIGINAL DESIGN LIGHT TENSION-SHEAR		16,600
REDESIGNED HEAVY TENSION-SHEAR		STRINGER 74,000 SKIN, 88,000
REDESIGNED LIGHT SHEAR		INITIAL 92,200 FINAL 111,520
REDESIGNED HEAVY SHEAR		INITIAL 99,000

* FATIGUE TESTS CONDUCTED UNDER REPRESENTATIVE FLIGHT-BY-FLIGHT FATIGUE TESTING

Fig. B-14

CORRELATION BETWEEN STRESS SEVERITY FACTOR AND FATIGUE

Fig. B-15

I-38
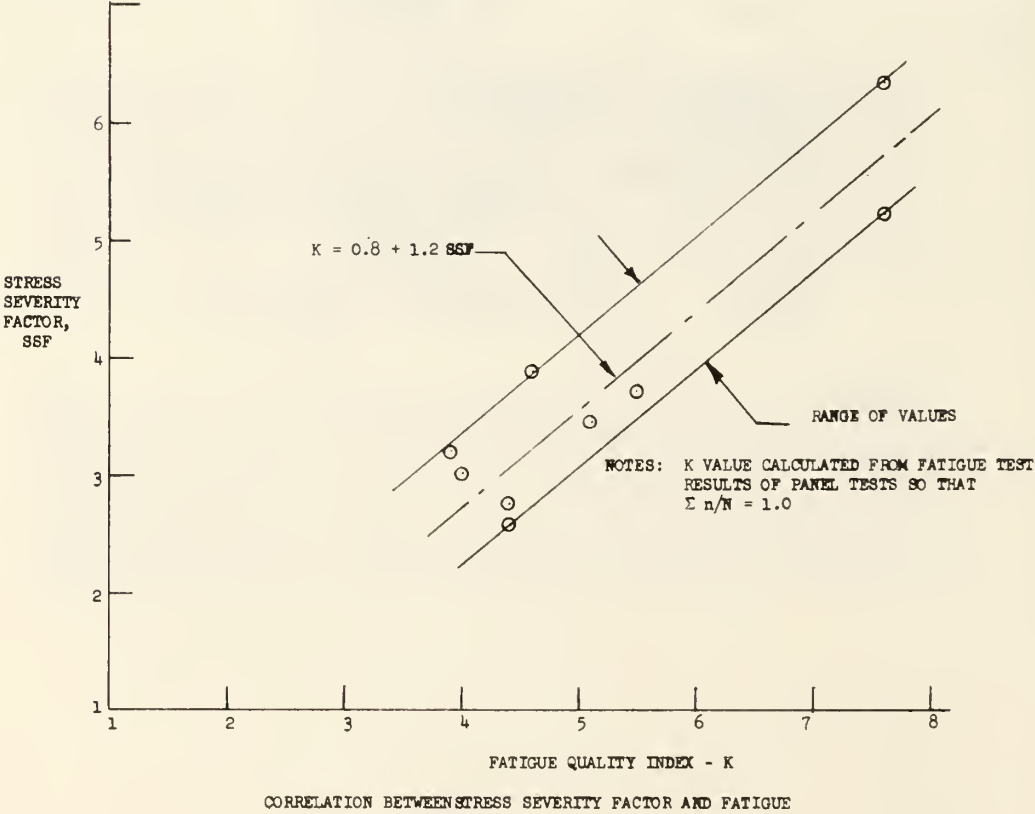
FRACTURE MECHANICS CONSIDERATIONS
IN DESIGN AND MANUFACTURE

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Abstract: Problems encountered on the F-111 in detail design and
inspection procedures are discussed; subsequent programs
toward establishing service life integrity and fracture
mechanics data are described; and methods for deter-
mination of inspection intervals and crack retardation
are given. The designer's problem is summarized as
shown in Fig. C-39, 40 -- emphasizing the need for
intimate association of the designer with manufacturing
and inspection operations.

As you are all well aware, there are many basic considerations
involved in structural design. These have been briefly summarized
and shown on Fig. C-1. I would like,however,to discuss two
specific items concerned with service life, Detail Design and
Manufacturing, as they have been responsible for most of the
difficulty in service life structural certification of the F-111,
and in fact, I believe are the major contributors to service
life problems on most aircraft.

Before discussing these items in detail, I believe it is
important to get a feel for the significance of the structural
problems which have occurred in military aircraft service usage
and to view them from the proper perspective. An examination
of the statistics covering a period of about eight recent years
indicates the incidents of inflight primary structural failures
of military aircraft as shown on Fig. C-2. Only a fraction of
the failures shown have resulted in loss of the aircraft but the
magnitude of the problem is evident. It is also apparent that
the majority of failures occurred in the wing structure.

Fig. C-3 shows the loss statistics for USAF fighters due
to all noncombat causes. An average yearly loss of 139 airplanes
translates to a little over 11 airplanes destroyed per 100,000
flight hours. More specifically the losses due to structural
failures have been about one per year or an average rate of
.103 per 100,000 flight hours, or just under 1% of the total
USAF noncombat fighter losses. Fig. C-4 presents the same
data by aircraft type.

I would now like to discuss a few of the problems which
occurred in the structural history of the F-111 during structural
certification and service usage. The most significant of these
have been associated with the high heat treat steel parts and

have been the result of the detail design or the fabrication and inspection aspects. Most of the problems, as shown on Fig. C-5, were isolated in the laboratory during full-scale fatigue tests or during the course of inspection and proof test of the airplane. However, there has been one inflight structural failure. You will note that our experience is in part similar to the experience of LTV described in an earlier presentation in that three of the problems were due to stress corrosion cracking, none of these were service induced.

There is a considerable amount of high heat treat steel used in flight critical parts on the F-111 airframe. The definition of critical in this case is a part or component essential to maintaining structural integrity of the primary flight structure. The total usage of steel represents approximately 20% of the structual weight of the airplane or about 5,000 pounds. As shown in Fig. C-6, the use of steel in critical parts is essentially concentrated in the heavily loaded structures, such as the wing carrythrough box, wing pivot fittings, tail support bulkhead and longerons. With few exceptions, the steel employed in the F-111 is D6AC, a derivative of an alloy which was initially developed by Ladish. It is similar in many of its characteristics to 4340 except it does not exhibit a blue brittle heat treat range. Extensive use is made of welding on the three major wing fittings; however, this has not been a source of any service or test problems.

The F-111 structural test program included a full-scale static test and a full-scale fatigue test article. Fig. C-7 shows the fatigue test article set-up for testing. The fatigue test originally was planned to be conducted, as shown on this chart, on a complete airplane including wing, fuselage and tail. As the program progressed it was revised to test individual major components to expedite the program. This was particularly feasible for the F-111 due to the nature of the wing attachment to the fuselage inherent in the variable sweep design.

The first failure in the airplane fatigue test program occurred in the wing carrythrough structure approximately $2\frac{1}{2}$ years ago, as shown on Fig. C-8. Approximately 1800 cycles of 4g load had been applied when failure occurred in this wing support structure. The origin of the failure was a taper-lok bolt which attaches an aluminum door to the rear spar structure. The crack at the point of unstable crack growth was approximately .2", and progressed across the sculptured lower plate of the box. In the bolt hole adjacent to the origin of the failure, it was discovered on post-mortem examination that a fatigue crack also existed.

In view of this early failure, an extensive investigation was undertaken, and it soon became obvious that there were two problems. As shown in the magnified pictures of Fig. C-9, it was found that some rather rough drilled and reamed holes

existed. The profilcorder measurements shown indicate the surface condition. Measured with a profilometer, the surface finish is RMS of 200 to 250. There was also concern about the stress levels in view of the evidence of fatigue in the adjacent hole to the failure origin, and a stress survey was conducted in the steel flange corner radius. Strain gages were installed, as shown on Fig. C-10, on the flange and slug area not only on a full-scale test specimen which was set up specifically for this purpose, but also on the static test airplane and on a flight test aircraft. With a nominal stress of approximately 80,000 psi in the slug area, a factor of about 2.5 times this value existed on the top edge of the flange.

The solution therefore involved both design and manufacture. Figure C-11 is a simplistic chart but it makes the point. With the high stress concentration as measured and the rough hole surface finish a low fatigue life is indicated. It was necessary to lower the stress level and to reduce the taper-lok bolt loads by design, and additionally, revise the manufacture technique to improve bolt hole finish and obtain better bolt fits to achieve a lower effective stress concentration.

As quite a large number of parts had already been manufactured, retrofit considerations were of prime importance. A considerable amount of analytical and test effort was required to develop a simple retrofitable fix that could be applied without causing other problems within the constraints of minimum rework to the high heat treat steel structure. The final design is shown in Fig. C-12 and represents the design correction aspect of the problem. The gusset design, though not evident from the chart, required precise sculpturing to obtain the optimum stress distribution necessary to reduce the flange stresses and to achieve minimum bolt loads. The significant reduction in stress level achieved is apparent from the chart.

At the same time and during the period of several months of testing of the design improvement change, process improvements were also being vigorously pursued. The resulting changes are shown on Fig. C-13, and involved the development of special multi-fluted carbide tapered reamers, and revised manufacturing procedures including such things as use of improved coolants for the reaming process. An extensive test program on effects on fatigue life of bolt interference fit was also conducted, and as a result the installation tolerances were considerably tightened.

These detail changes in design and manufacture were subsequently proven by subjecting a full-scale test specimen to the equivalent of 24,000 flight hours or 6 service lives without failure.

As noted previously, the fatigue test program was revised to test major components, and Fig. C-14 shows the wing in the fatigue test fixture. The F-111 contractual requirements are

that the airplane be designed and tested to four lives. At just over three lives in the initial test of the wing a failure occurred at the point shown in Fig. C-15. Figure C-16 is a photograph of a production part with the failure line indicated. The failure initiated in the lower plate of the fitting at one of the fuel flow holes machined in the integral ribs.

Figure C-17 shows the fractured part at the point of failure origin. The test had been run in repeated spectrum loaded blocks so the block markers were quite clear and permitted the determination of the crack growth rate. The critical crack depth for this part is about $\frac{1}{4}$" which results in a surface crack length of approximately $\frac{1}{2}$". We were well aware that these fuel flow holes did represent a stress concentration and this area was used as a fatigue analysis control point. Moreover, a number of element tests of the part had previously been run satisfactorily to specification requirements. It was determined on examination of these original test specimens and the full-scale wing test that the surface finish of the test specimens was better than the fatigue test full-scale wing.

When this part was initially designed, we were not as adept at using finite element analysis as we have since become and subsequently, a fine grid finite element analysis was made of this area as depicted in Fig. C-18. The average stress field of 103,000 psi, used as a reference, corresponds to the maximum stress expected in the fatigue test spectrum. The close correlation of the calculated values to measured values, which were taken from strain measurement of a full-scale part is evident. The origination of the fatigue crack was at the left lower corner of the fuel flow hole where the stress levels indicate approximately 190 ksi.

Again, the problem existed of having a large number of parts manufactured and installed on airplanes which were flying. While this particular failure represented no immediate service problem, in view of the relatively long demonstrated life, improvement was required. Figure C-19 shows the application of an epoxy boron reinforcement on this steel plate. It was fortunate that the outside surface of the fitting was relatively flat without any ribs or stiffeners thus providing a good surface for bonding of the composite. Through development test and analyses, a basic design and orientation of boron laminates was selected which was most compatible with the modulus of elasticity of the steel, and provided acceptable bond shear loads. The objective of this reinforcement was fatigue enhancement, not static load since static testing had been successfully completed, so the design was tailored to specifically accomplish the objective of reducing the stress levels at the fatigue critical point. Figure C-20 is a photograph of this somewhat unique application of filamentary composites as a reinforcement. It is quite easily applied in production in an autoclave, and can be installed on complete

wings with use of portable tools employing electric heaters and pressure bags. The reinforcement weighs 6 pounds, and by analysis essentially doubles the fatigue life of the part. This design has been tested on a complete full scale wing through five lives without evidence of any fatigue distress.

Figure C-21 illustrates the effectiveness of the reinforcement. The values are measured stress levels from a full scale test wing and show the significant reductions and uniformity of stress level achieved. The finish of the rib intersections with the plate were also improved.

I would like to touch on one other problem which resulted in a flight failure. Figure C-22 indicates a manufacturing defect in a wing pivot fitting which escaped detection during manufacture. The small light zone at the base of the black area is the only region showing evidence of fatigue. Failure occurred during a 4g maneuver after a relatively short time in service. While we had been concerned in the past with the problem of fracture of these high strength materials, as a result of the aforementioned test experience, a rigorous fracture mechanics test and analytical approach had not been applied. As a result of this flight failure an extensive fracture mechanics program was initiated and has progressed a long way toward understanding and quantifying crack propagation and brittle fracture phenomenon. Some of these actions taken on the F-111 will undoubtably benefit and be reflected in future programs.

Figure C-23 shows the elements which essentially constitute the service life integrity program for the F-111 as modified to include fracture mechanics considerations. Involved in quantifying the Fracture Mechanics inspection interval calculations is the prediction of crack growth rates and critical crack sizes as indicated. This required a considerable test and analysis program as an adequate test data base and analytical procedures to accomplish these analyses did not exist. The basic elements of this program are shown in Figure C-24. Essentially, the procedure starts with the service usage spectrum used for fatigue analysis with the additional consideration of the temperature and chemical environment. As it is not practical to perform an analysis for every part of the airplane, it was necessary to isolate the critical parts which, if they fail, would cause catastrophic failures. In some instances the criticality of parts could be determined by simple examination. However, in a large number of cases extensive failure analyses were required. Flaw growth models were also required that were representative of the structure and correlated with spectrum test. It became apparent that there was a considerable difference between spectrum loading and constant amplitude cyclic loadings on crack propagation in steel and this type testing was included in the specimen test program. Another unique procedure adopted for the F-111 was to proof test airplanes on a production basis and

this was made an integral part of the fracture control program. Also, a statistical risk assessment analysis was performed which was useful in establishing the relative criticality of important structural part.

Figure C-25 illustrates the basic proof test concept. The premise is that the test will demonstrate that a crack greater than a critical depth will not exist at a given stress level and temperature. Further, in actual operation, the airplane will in all probability experience some lesser stress level and at a higher temperature than the proof test values. Advantage can be taken of both the lower stress and the fact that steel parts have a considerably reduced fracture toughness level at a low temperature to provide a margin as illustrated which can be converted to flight hours for a crack, if present, to propagate to critical size. In essence, the procedure is simple however there are a considerable number of problems involved in the practical execution. Figure C-26 is a superimposed photograph of an F-111 airplane subjected to positive and negative loads in the cold test chamber. As previously mentioned, the assumption is that any cracks which may exist in the structure are below their critical crack size for the conditions tested or catastrophic failure would result.

The basic fracture mechanics data program conducted for the F-111 involved not only Convair but also a number of other test laboratories participated, including those at Boeing, Battelle, Aerospace Corp., NASA, AFML and AFFDL. A large number of tests were required to isolate the variables shown in Fig. C-27. Close to a thousand specimens of D6AC 220-240 heat treat steel were run in this basic data program. Figures C-28 to C-30 illustrate some of the types of specimens and typical test machines which were used to obtain this data.

An unforeseen difficulty was encountered in establishing the variation of K_{1c} with temperature which is required to permit prediction of critical crack size. It was found that D6AC did not have a single K_{1c} variation with temperature as illustrated by Fig. C-31. This departure from the expected situation was determined to be a function of the cooling rate during the heat treat quench from a material temperature of 600°F down. Values of K_{1c} at room temperature were found to vary from 40 to 95 depending on this quench rate. It was therefore necessary to determine the heat treat histories of all the parts installed in the completed aircraft, and to take into account the fracture toughness value that would produce the shortest or most conservative inspection interval. It should be noted that the other material properties: impact strength, elongation, yield strength and ultimate strength, were within specifications values and did exhibit this variation. Also, in spite of the variability of fracture toughness no difference is discernible in crack propagation rates.

The data plotted on Fig. C-32 is typical of the data derived from the test program and illustrates the relative influence of environment on crack growth for constant amplitude WOL type specimens. The Δk range for the data presented is of the order of 10 to 60. Similar data from the complete test program have been assembled and will be published shortly by the Air Force Material Lab.

As previously noted, it became apparent during the course of the program and from evaluations of data available that the constant amplitude test specimen program data was not consistent with results from spectrum tests. A considerable retardation effect was evident from these spectrum loaded tests depending upon the order of the applied loading and environment. To resolve this difference an extensive spectrum/environment effects program was initiated as summarized in Fig. C-33. A baseline program was run with certain conditions held constant as noted and variations evaluated against this baseline. There was a total of 109 specimens in this program and again a number of laboratories participated in addition to General Dynamics. Figures C-34 and C-35 show some of the typical test set-ups used.

To establish an analytical correlation, a mathematical model was developed by Dr. Wheeler of Convair, as shown on Fig. C-36. This model accounts for the crack retardation which takes place when a large initial load is followed by subsequent smaller loads. The derivation of this equation has been published and is available in the engineering literature. The significance of the retardation effect is shown in Fig. C-37. The particular test illustrated was run to a 5g maximum load and a mission developed spectrum in a JP-4 fuel environment. The spectrum loads were applied in 58 random load levels. An analysis based on constant amplitude test data would predict that the crack would propagate to a .20" depth after 400 hours. The specimen as tested took 3,200 hours to progress to this depth. A prediction based on using the retardation approach previously discussed is shown for comparison.

I would now like to discuss one of the specific aspects of the design problem today. In the past, fatigue analyses were performed and tests conducted on the assumption that the service life was a function of the time required to incubate and initiate a crack and to have it progress to failure. The current concept interjecting Fracture Mechanics starts with the assumption that there is a pre-existing crack. Figure C-38 is an artist's rendition but it illustrates the obvious fact that the left-hand curves will produce an appreciably shorter life than the right-hand curve. It is also generally recognized, as noted on the chart, that there are a large number of factors involved in the development and growth of flaws. The net result is that the design task, if it is to take into account Fracture Mechanics considerations directed toward specific requirements, will be considerably more complex.

In conclusion, I have summarized on Figures C-39 and C-40 the Designers problem as it can be viewed today in regard to achieving adequate and safe service life. To cope with these problems, new methods and procedures need to be developed and integrated into the design process.

BASIC STRUCTURAL DESIGN CONSIDERATIONS

- STATIC STRENGTH

- FLUTTER/AEROELASTICITY

- SERVICE LIFE

  - APPLIED LOADS (Magnitude/Spectrum)

  - INTERNAL LOAD DISTRIBUTION

  - DETAIL DESIGN (Concept, Analytical Methods, Materials)

  - MANUFACTURING (Fabrication & Inspection)

  - SERVICE USAGE (Operational, Maintenance)

FW7-274-V9267F
'3 JUL 71

Fig. C-1

MILITARY AIRCRAFT IN-FLIGHT FAILURE OF PRIMARY STRUCTURE

WING FAILURES				FUSELAGE / TAIL	
F-4	(8)	B-57	(2)	F-101	(1)
F-5	(1)	C-5	(1)	F-105	(3)
F-8	(6)	C-124	(1)	B-52	(6)
F-84	(2)	C-133	(1)	C-47	(1)
F-86	(1)	T-33	(1)	C-130	(1)
F-100	(3)	T-37	(1)	C-133	(1)
F-102	(3)	A-1	(2)	KC-135	(1)
F-104	(1)	A-4	(2)	T-33	(1)
B-26	(1)	A-6	(1)	T-37	(1)
B-52	(3)	F-111	(1)		
	TOTAL - (42)			TOTAL - (16)	

- ABOVE FAILURES OCCURRED DURING 1962 THROUGH SEPTEMBER 1970 TIME PERIOD EXCEPT FOR B-52 AND B-57 FAILURES.

- B-52 AND B-57 FAILURES OCCURRED DURING 1959 THROUGH 1969 TIME PERIOD

- FAILURES INVOLVE BOTH USAF AND USN AIRCRAFT AND IN SOME CASES RESULTED IN LOSS OF AIRPLANE.

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Fig. C-2

USAF NON-COMBAT FIGHTER EXPERIENCE

1962 - 1969 INCLUSIVE*

- USAF FIGHTERS DESTROYED - ALL NON-COMBAT CAUSES
  - Average Losses per Year 138.875
  - Average Rate per 100,000 Flight Hours 11.47

- USAF FIGHTERS DESTROYED - FAILURE OF PRIMARY STRUCTURE
  - Average Losses per Year 1.25
  - Average Rate per 100,000 Flight Hours 0.103
  - Portion of Total Fighter Losses Caused by Failure
    of Primary Structure 0.9%

- FAILURE OF PRIMARY STRUCTURE IS CAUSE OF ABOUT 1% OF ALL USAF NON-COMBAT
  FIGHTER LOSSES

NORTON AFB DOC. AS-X16 INCLUDES F-4, F-5, F-86, F-100,
F-101, F-102, F-104, F-105, F-106, & F-111A/E/D/F

FWT: 274-152083
13 JULY 71

Fig. C-3

USAF FIGHTERS (Non-Combat)

1962 - 1969 INCLUSIVE*

AIRCRAFT	LOSSES ALL CAUSES	LOSSES-FAILURE OF PRIMARY STRUCTURE	TOT. FLIGHT HOURS	LOSS RATE ALL CAUSES (Per 100,000 Flt Hrs)	LOSS RATE FAILURE OF PRIMARY STRUCTURE (Per 100,000 Flt Hrs.)
F-4	167	0	2,163,094	7.71	0
F-5	9	0	45,787	**	**
F-86	44	1	231,713	18.99	.432
F-100	328	3	2,543,027	12.90	.118
F-101	99	1	1,122,795	8.82	.089
F-102	132	1	1,572,305	8.40	.064
F-104	82	1	332,016	24.70	.301
F-105	193	2	1,088,519	17.73	.184
F-106	47	0	538,278	8.73	0
F-111A/E/D/F	10	1	85,075***	**	**
Total	1,111	10	9,689,622	11.47	.103

*Norton AFB Doc. AS-X16
**Not Comparable Due to Program Stage
***As of 2 July 1971

FWT: 274-152083
13 JUL 71

Fig. C-4

F-111 STRUCTURAL HISTORY

- MOST SIGNIFICANT PROBLEMS HAVE BEEN ASSOCIATED WITH HIGH HEAT TREAT STEEL

    - DETAIL DESIGN

    - FABRICATION INSPECTION

- SOURCE OF PROBLEM IDENTIFICATION

    - FATIGUE TEST (4)

    - SERVICE USAGE (1)

    - INSPECTION/PROOF TEST (3 S.C.C. 1 Manufacture)

Fig. C-5

CRITICAL STEEL PARTS

STABILIZER HORN RIB

RUDDER TORQUE TUBE

UPPER LONGERONS

WING PIVOT FITTING (WPF)

WING PIVOT PIN

770 BULKHEAD

496 NACELLE FORMER

WING CARRY THROUGH BOX (WCTB)
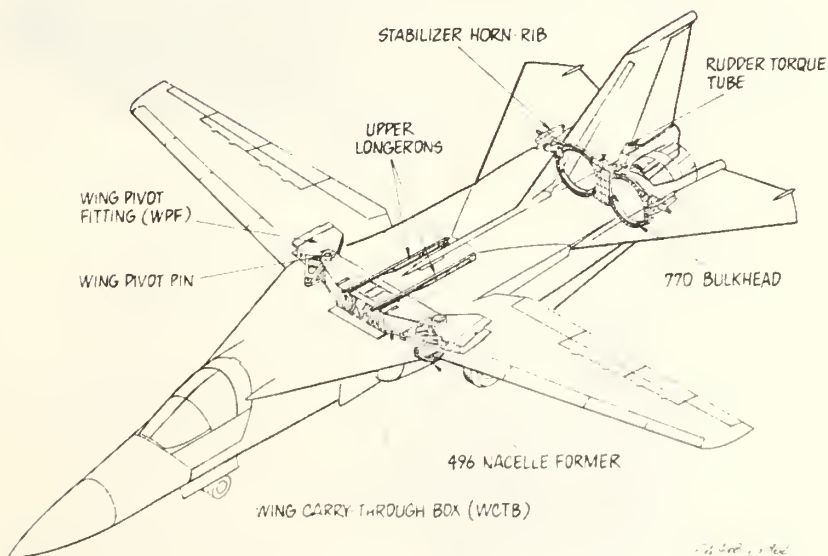
Fig. C-6
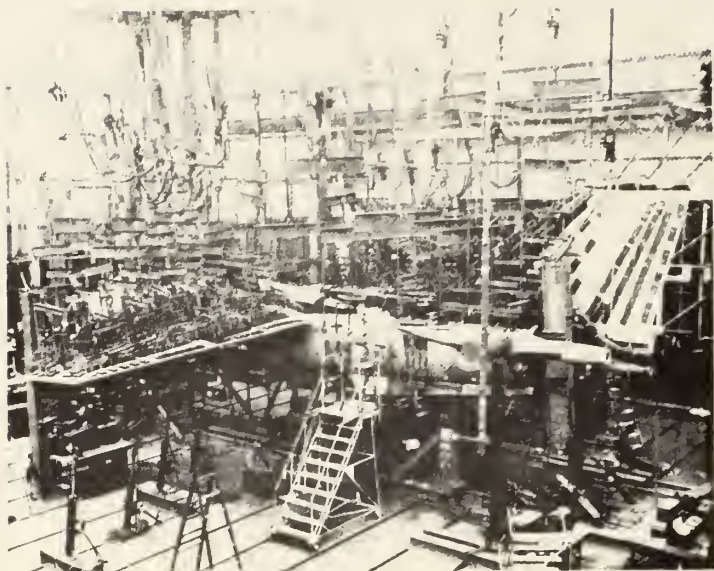
Fig. C-7

A-4 FAILURE - CARRY THRU BOX
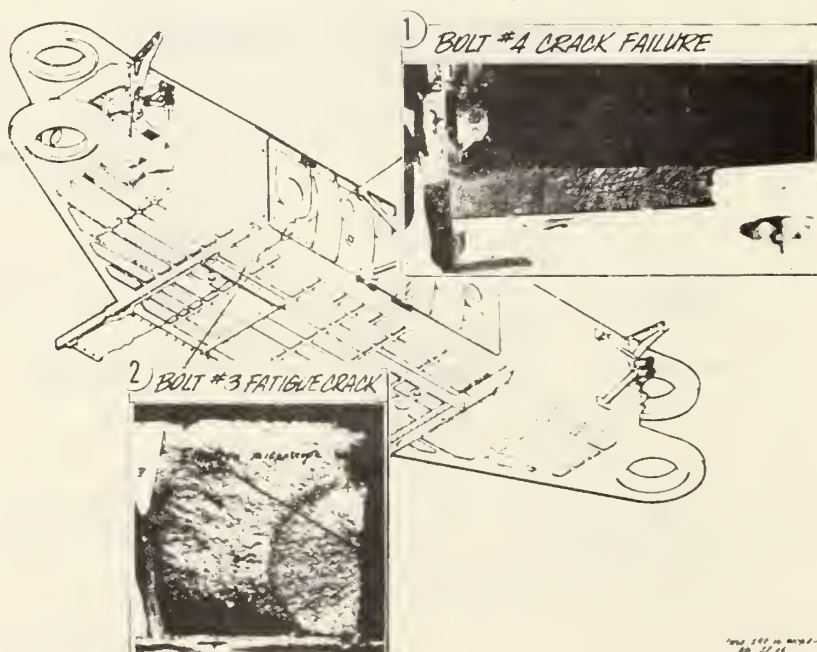
Fig. C-8

CONDITION OF HOLES IN A4

HOLE # 3   HOLE #4

PROFILCORDER MEASUREMENTS

HOLE #4

HOLE #3
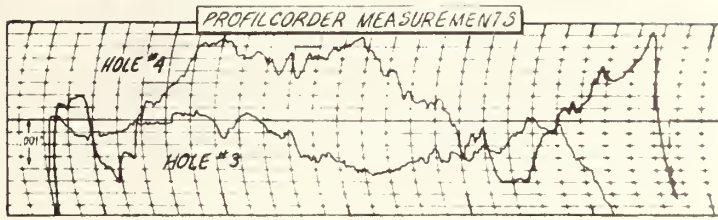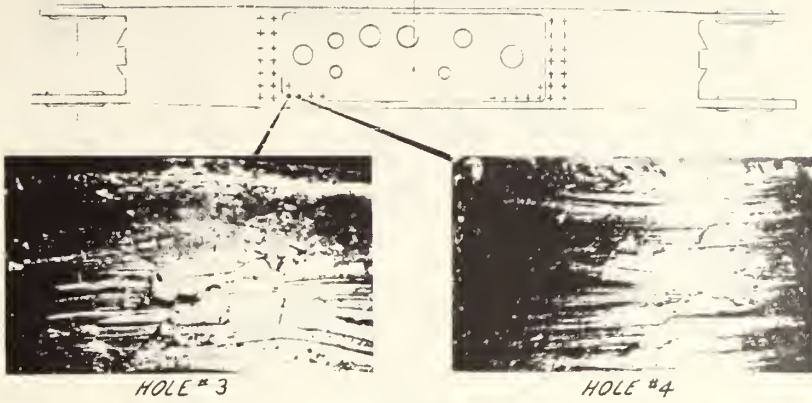
SURFACE CONDITION OF HOLE No 3 & No 4 FOUND TO BE VERY ROUGH

Fig. C-9

CORRELATION OF CORNER STRESSES

FW #1   ○ ———
A-3 (Static test)   △ — — —
AIRPLANE #23   ☆ - - - - -

UPPER
FLANGE
BELOW
BOLTS
SLUG

UPPER FLANGE STRESSES

BELOW BOLT STRESSES

SLUG STRESSES

STRESS - KSI
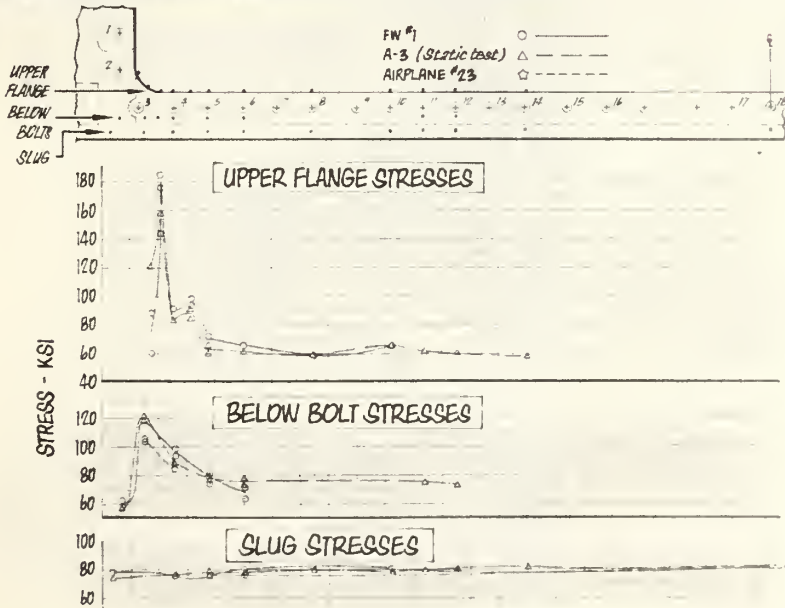
Fig. C-10

FATIGUE DESIGN FACTORS
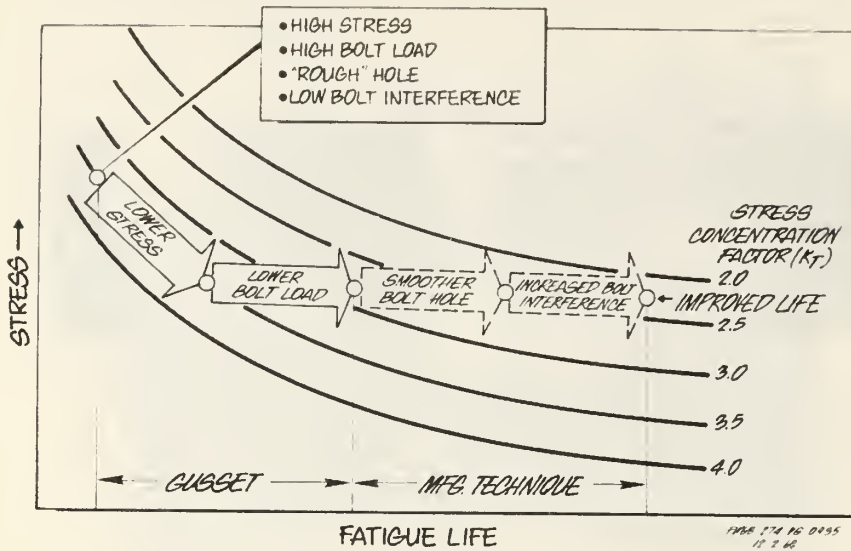
Fig. C-11

EFFECT OF MOD I MINI GUSSET (-568)
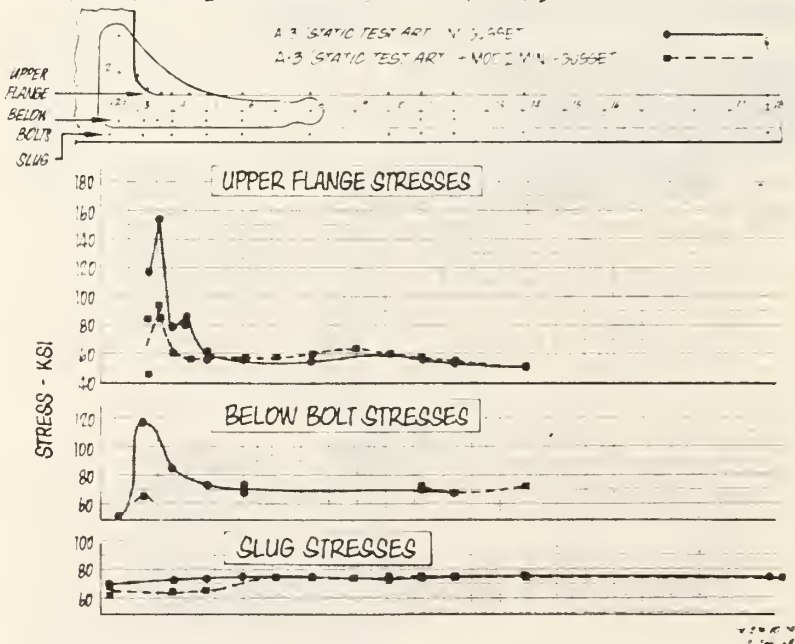
Fig. C-12

PROCESS IMPROVEMENTS

- THREE-STAGE REAMING SEQUENCE
- IMPROVED COOLANT FOR FINISH REAMING
- POWER FEED TAPER REAMING
- 18 FLUTE CARBIDE FINISHING REAMER — 15 PMS
- INCREASED PROTRUSION OF TAPER LOK BOLTS

Fig. C-13

A-4 FATIGUE TEST WING
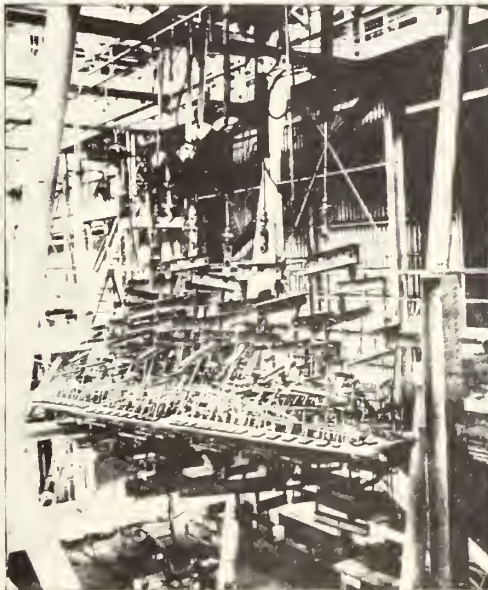
Fig. C-14

F-III WING ASSEMBLY-LOWER SURFACE

Fig. C-15

LOCATION OF R.H. WING-FAILURE
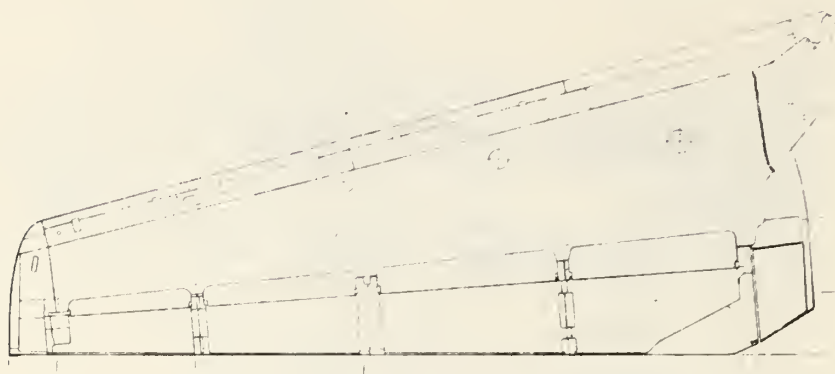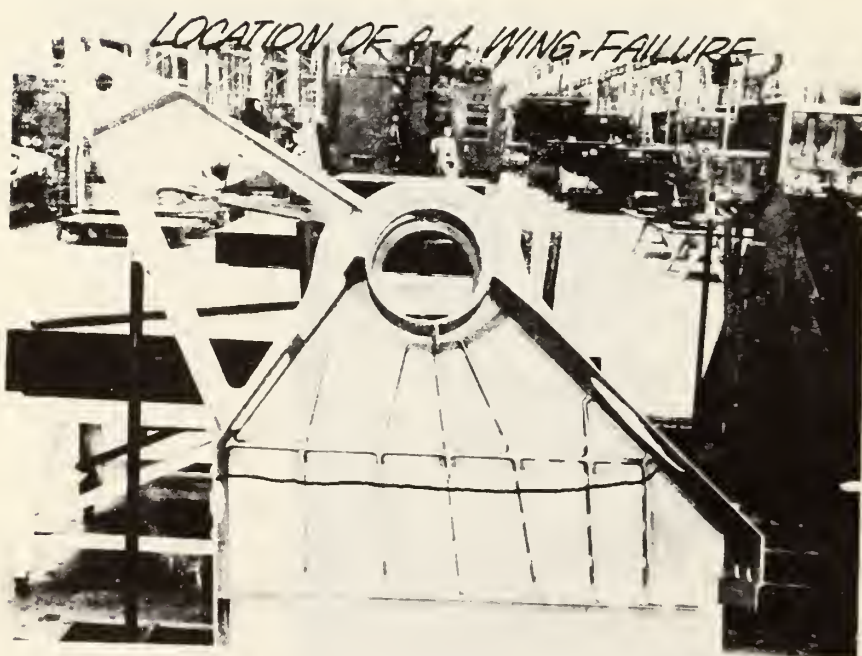
Fig. C-16

CRACK PROPAGATION A-4 FUEL FLOW HOLE
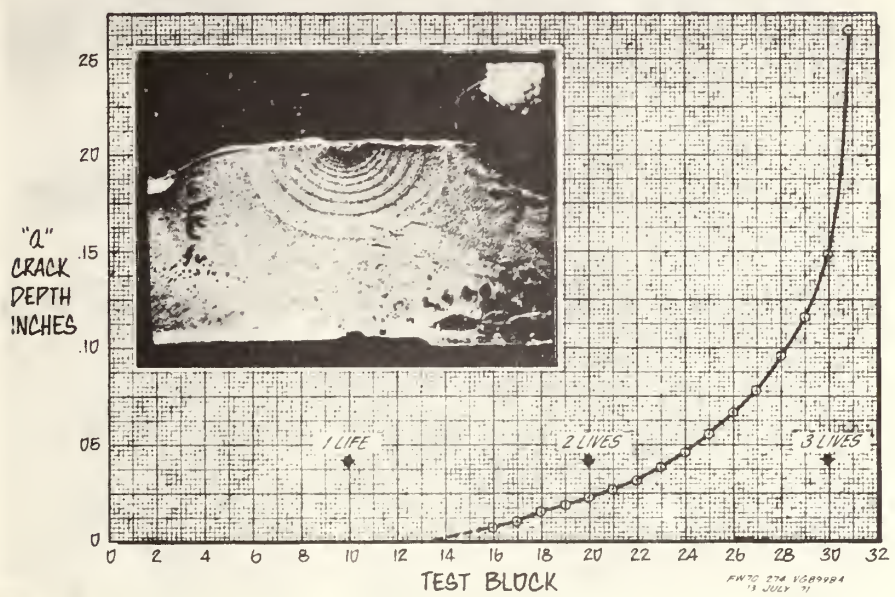
Fig. C-17

FINE GRID FINITE ELEMENT ANALYSIS
FUEL FLOW HOLE NO. 4

NOTE:
ALL STRESSES (KSI) ARE
FOR MAX. FATIGUE LOAD
* FIGURES FROM A-4
STRAIN SURVEY

AVG. FIELD
STRESS
THIS AREA
IS 103

90

194
185 * 225
242

71
191 *
194

69 65
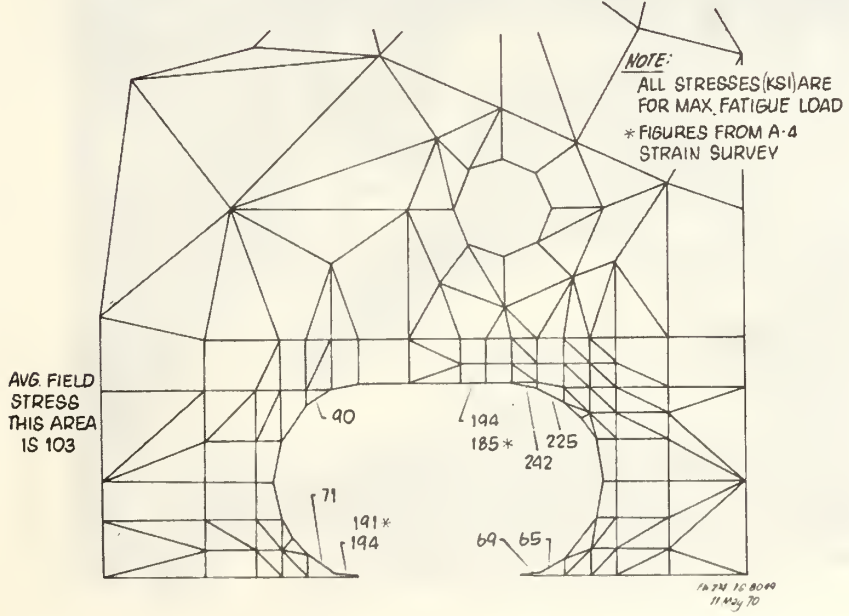
Fig. C-18

FINAL CONFIGURATION OF BORON DOUBLER
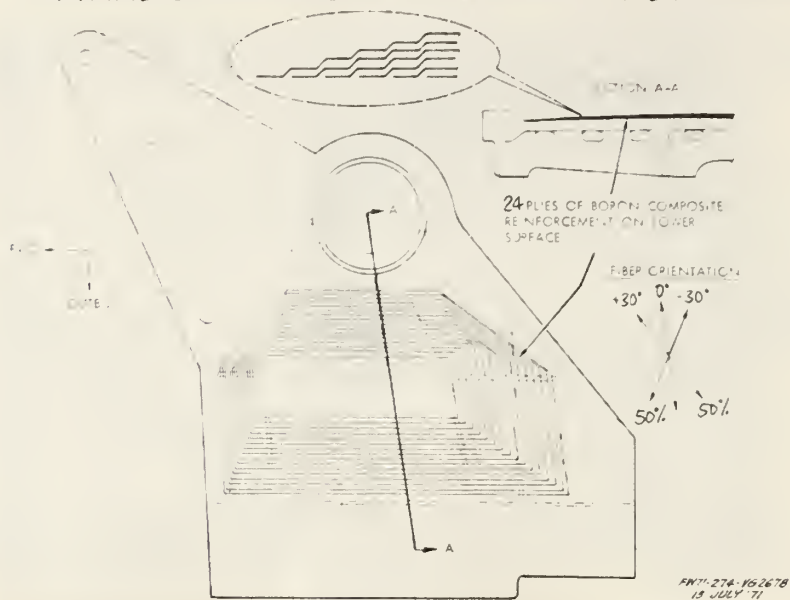
Fig. C-19
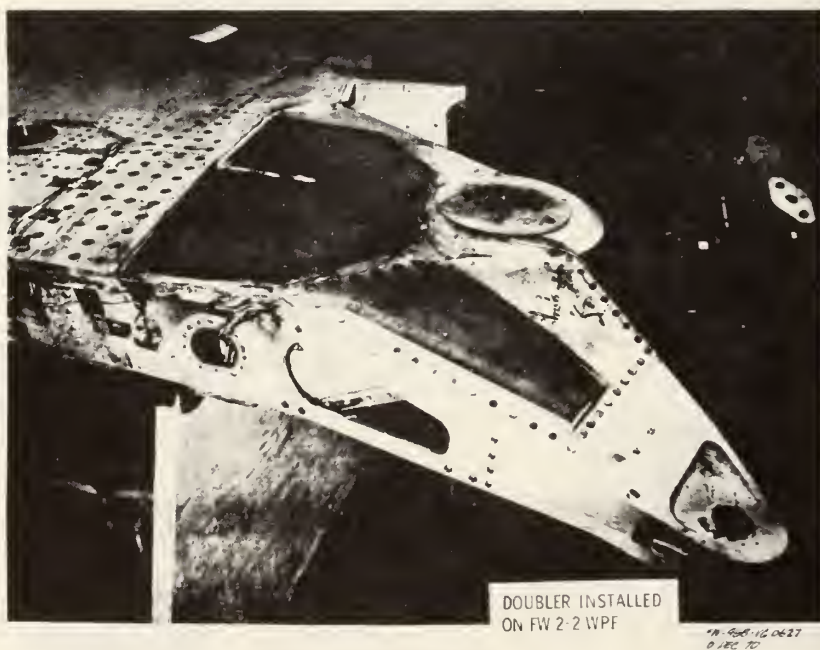
Fig. C-20

WING PIVOT FITTING – (CENTER SPAR AREA)
STRESS COMPARISON BEFORE & AFTER BORON REINFORCEMENT

BASIC	(74.0)?	114	174	126	700	657
WITH BORON	92.1	104	149	105	OUT	OUT

55 56 57 58 59 60 61 INB'D

BASIC	109	124	142	151	1.9	104	62.0
WITH BORON	107	114	110	109	116	96.8	56.3

NOTE STRESSES SHOWN ARE MAXIMUM STRESSES IN KSI FOR FATIGUE SPECTRUM
(EXTRAPOLATED FROM APPROX 50% OF LIMIT TO APPROX. 75% OF LIMIT)

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9 NOV 1970

Fig. C-21

LEFT WING-PIVOT FITTING ANOMALY

OUTBOARD PART - STAYED WITH WING

INBOARD PART - STAYED WITH AIRPLANE

AREA OF ANOMALY

.905   .236

STEEL .282 NOMINAL

BONDED ALUMINUM HONEYCOMB

ANOMALY IN STEEL FRACTURE

FW-292-16 7985C
28 DEC 970

Fig. C-22

SERVICE LIFE INTEGRITY
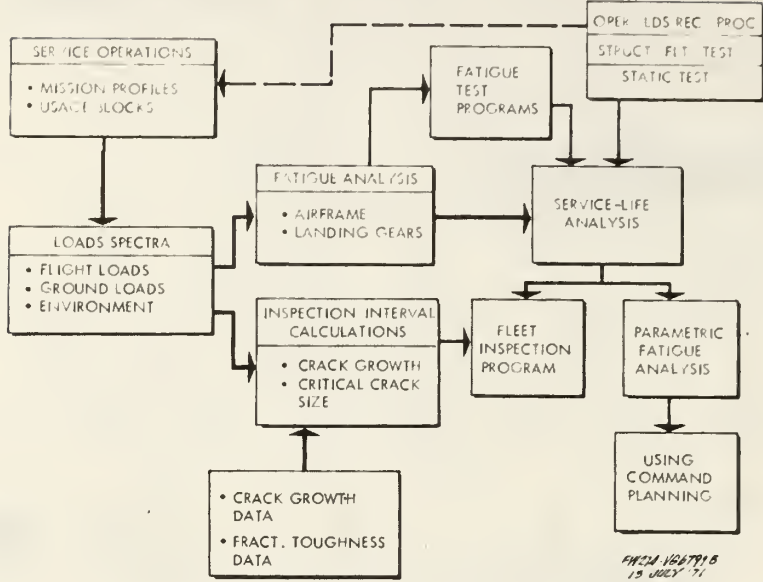
Fig. C-23
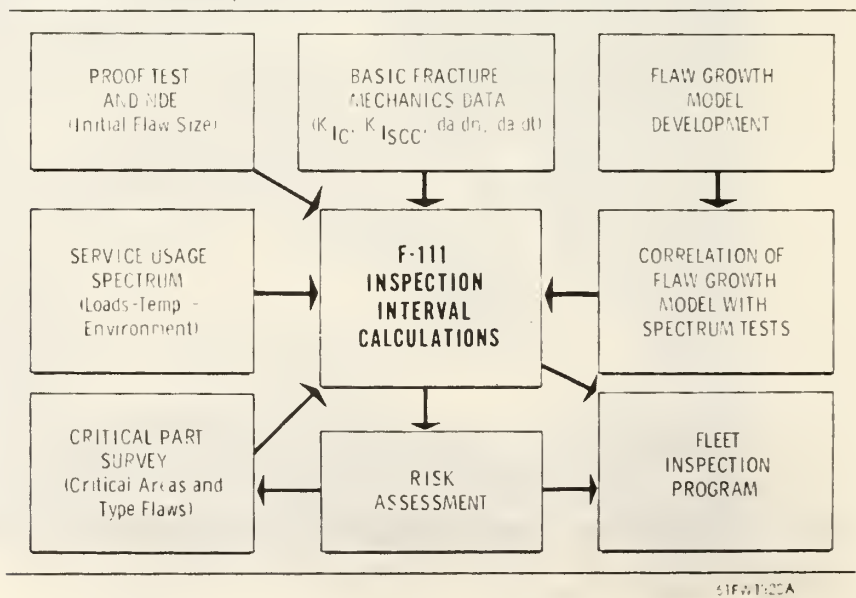
Fig. C-24

PROOF TEST CONCEPT
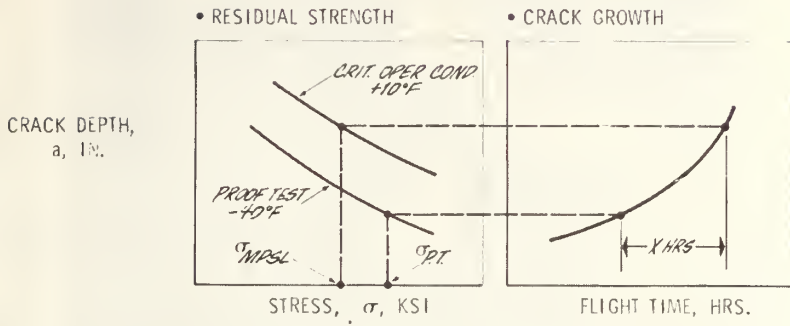
Fig. C-25
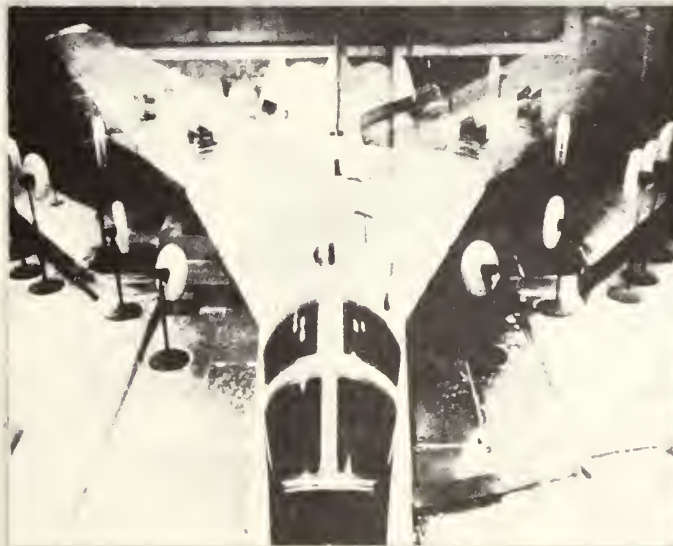
Fig. C-26

*VARIABLES INVESTIGATED	TYPE DATA			
	K_{IC}	K_{Iscc} & da/dt	da/dN	M_K
SPECIMENS: (CT SF DCB)	√	√	√	
MATERIAL: (Plate, Forging, Prod. Parts)	√	√	√	√
HEAT TREAT PROCESS: (Temperatures, Quench Rate, Quenchant, Circulation)	√	√	√	
THICKNESS: (0 25 to 0 75 in)	√	√	√	√
DIRECTIONALITY: (RT, RW, WR, TR, WT, TW)	√	√	√	
TEMPERATURE: (-65°F to 300°F)	√	√	√	√
ENVIRONMENTS: (Dry Air, Lab Air, JP-4, Distilled Water, 3-1/2% NaCl, Prussian Blue, Relative Humidity)		√	√	
FREQUENCY: (1,6, 60, 180, 600 cpm)			√	
LOAD RATIO: (0.1, 0.3, 0.5)			√	
DEPTH TO LENGTH: (a/2c = .09 to .56)				√
DEPTH TO THICKNESS: (a/t = .29 to .99)				√
*NO. SPECIMENS (GD/TOTAL): (722/930)	462/556	98/124	129/217	33/33

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Fig. C-27

TYPICAL FRACTURE MECHANICS SPECIMENS
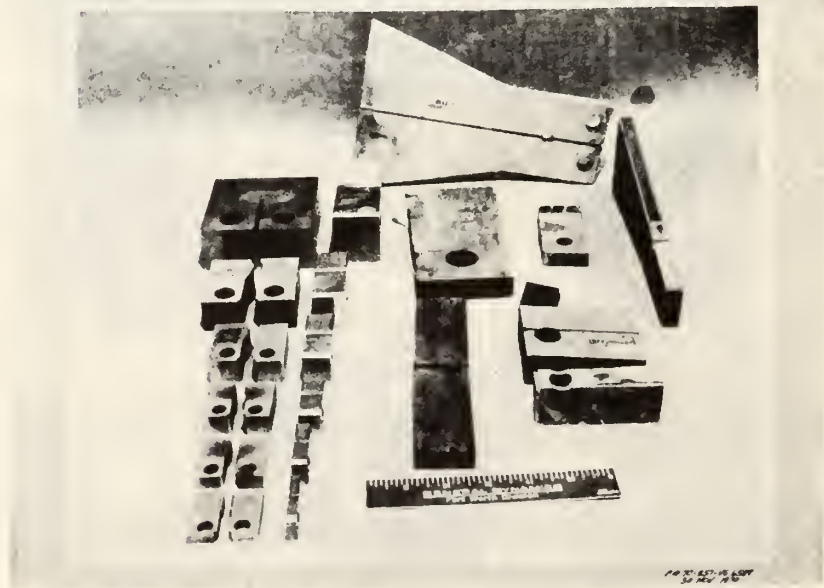
Fig. C-28

TYPICAL SUSTAINED LOAD TEST SET-UP

Fig. C-29

TYPICAL da/dN TEST SET-UP

Fig. C-30

K_{IC} VS. TEMPERATURE
Dbac, 220/240 KSI
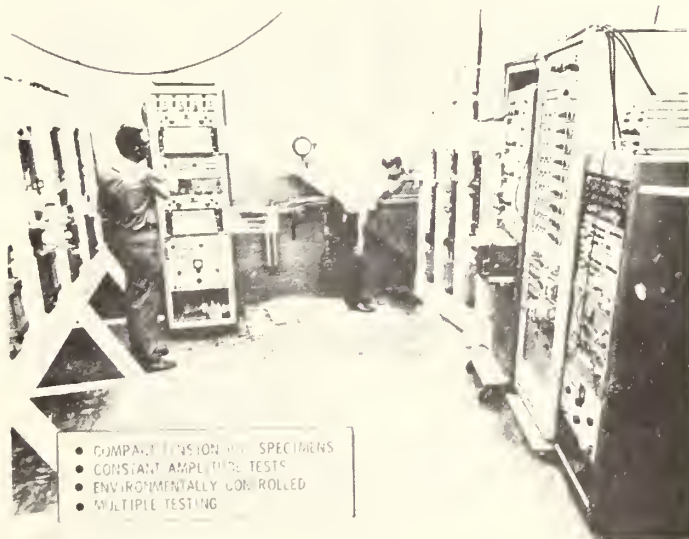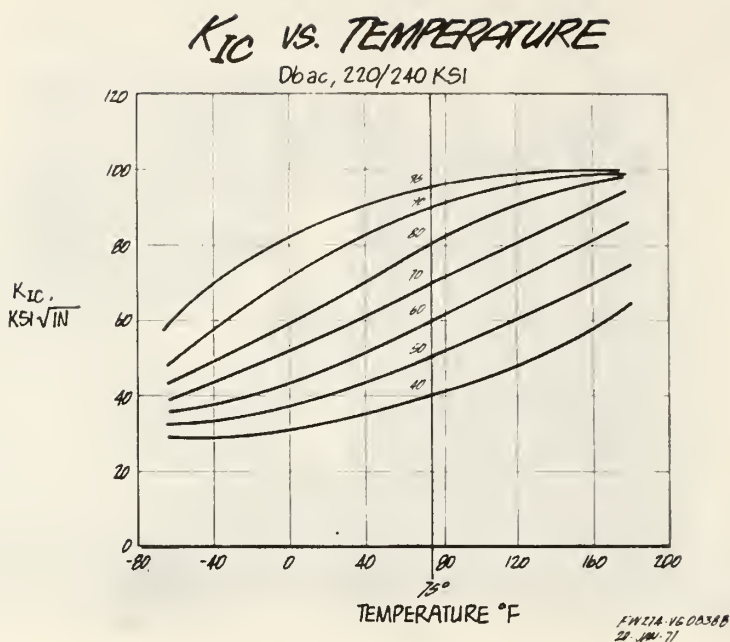
Fig. C-31

BASIC DATA-CRACK GROWTH IN VARIOUS ENVIRONMENTS
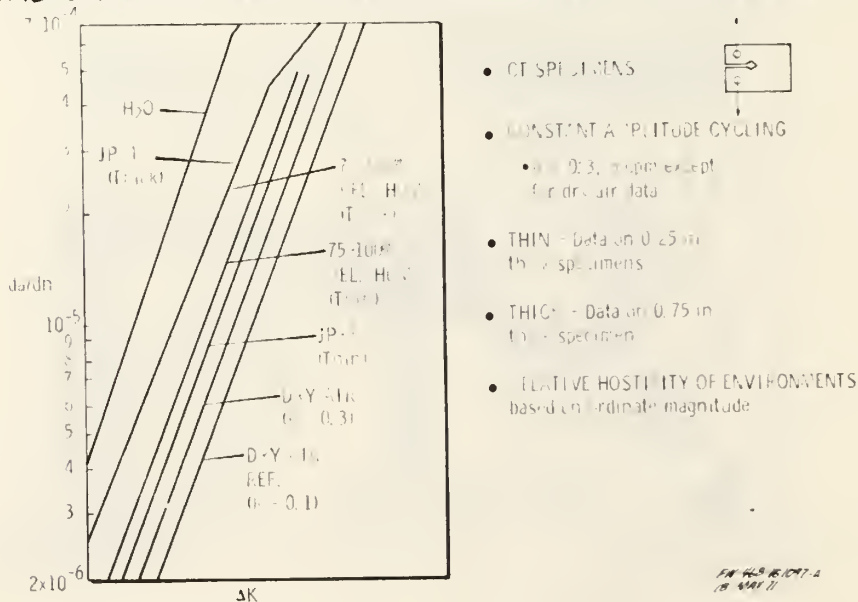
Fig. C-32

APPROACH

- ESTABLISH BASELINE DATA

    - CONSTANTS: K_{IC}, Flaw, Spectrum, Material Form, Thickness, Unit Stress
    - 4 ENVIRONMENTS: Dry Air, Rel. Hum., JP-4 Fuel, Dist. Water
    - 2 or More Crack Growth Curves/Environment

- EVALUATE EFFECTS OF:

    - SUSTAINED LOADS
    - COMPRESSION LOADS
    - FRACTURE TOUGHNESS
    - MATERIAL FORM
    - SPECIMEN THICKNESS

    - STRESS LEVEL (Unit Stress)
    - SPECTRUM SHAPE
    - LOAD SEQUENCE
    - SURFACE FLAW SHAPE
    - FLAW TYPE

NUMBER OF TESTS

- GD SPECIMENS / FLAWS _____ 48/77
- TOTAL SPECIMENS _____ 109

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Fig. C-33

SPECTRUM TEST CONTROL INSTRUMENTATION

- CRACK LENGTH MEASUREMENTS
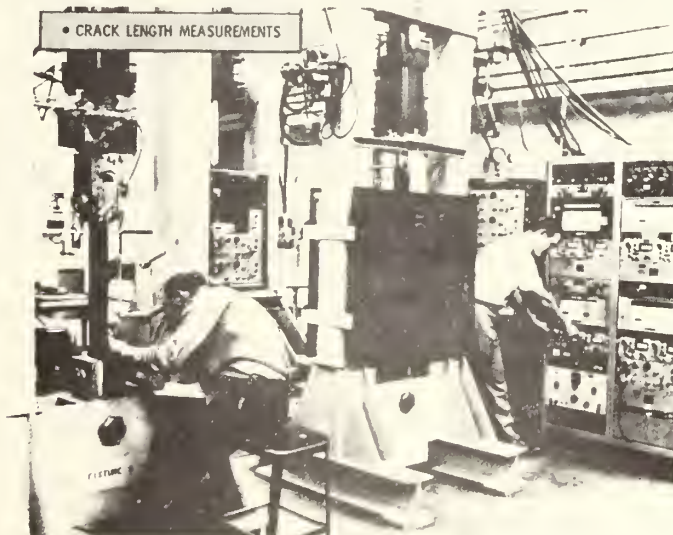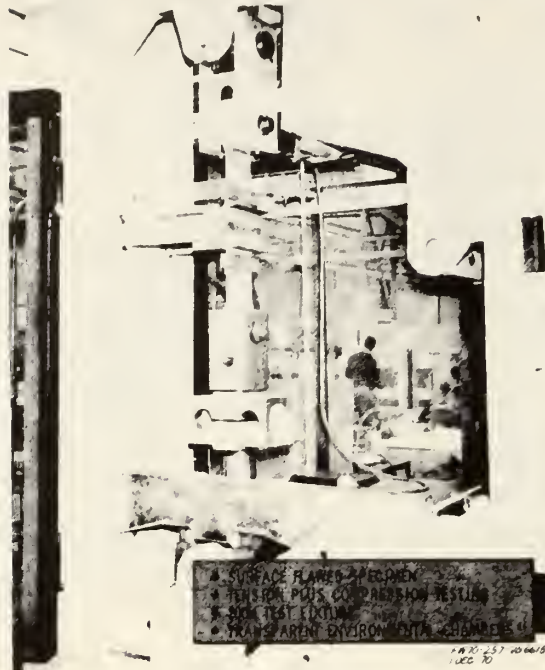
Fig. C-34

Fig. C-35

ANALYTICAL APPROACH

- $a + R_y \geq a_p$
  $C_p = 1$

- $a + R_y < a_p$
  $C_p = \left(\dfrac{R_y}{a_p - a}\right)^m$

- $\dfrac{da}{dN} = f(\Delta K)$

- $a = a_0 + \int C_p f(\Delta K) dN$

Fig. C-36

I-64
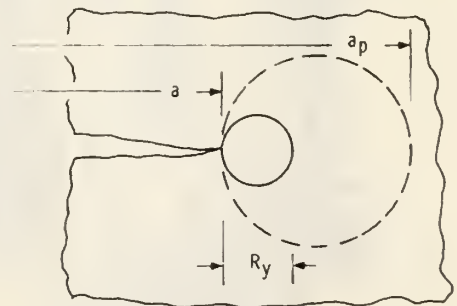
SURFACE FLAWED SPECIMEN #1 CRACK GROWTH vs. PREDICTED

TEST PLAN:
- 5.0 G MAC SPECTRUM
- 58 LOAD LEVELS
- RANDOM SEQUENCE OF LOAD LEVELS
- JP-4 FUEL 100% TIME
- NO PROOF TEST

$\alpha/2c = .5$

$\alpha_{cl} = .1979"$
$f_{CR} = 93.8$ KSI

$\alpha_{Initial} = .1204"$

CRACK DEPTH, IN.

NO. OF 200 HOUR TEST BLOCKS

TEST
CALC. "m" = 2.6
CALC. "m" = d (NO RETARD)

FW 274 V6 0739
2 DEC 1970
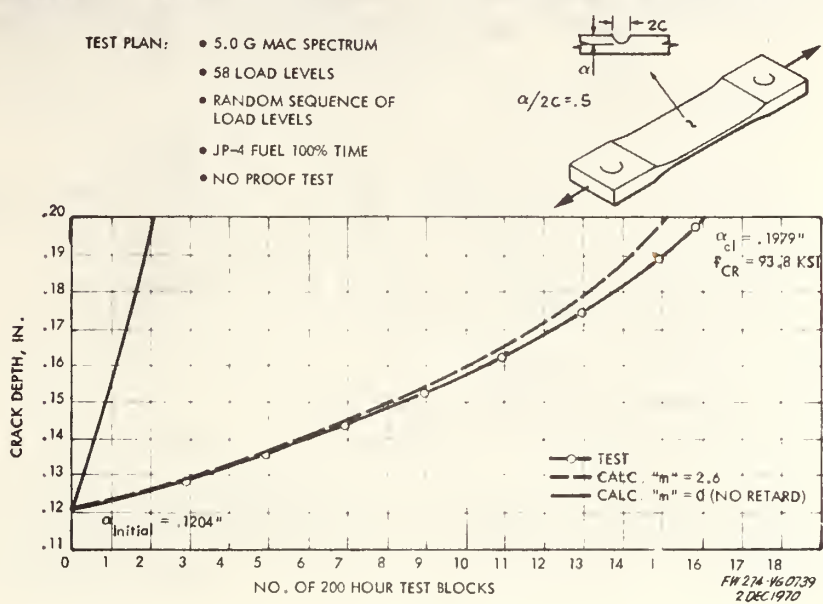
Fig. C-37

FLAW DEVELOPMENT STAGE

SCC

III

CRACK SIZE

CRACK SHARPENING

I — II — III

LIFE

STAGE		PRINCIPAL FACTORS
I	INCUBATION	A B-C-D E
II	MICRO-MACRO CRACK DEVELOPMENT	A-B-C
III	CRACK PROPAGATION	A-B-C-E

FACTORS

A - INTRINSIC MATERIAL PROPERTIES
B - DESIGN STRESS LEVELS LOAD SPECTRUM
C - DETAIL DESIGN

D - MANUFACTURING PROCESSES
E - ENVIRONMENT

FW 274 V8 2674
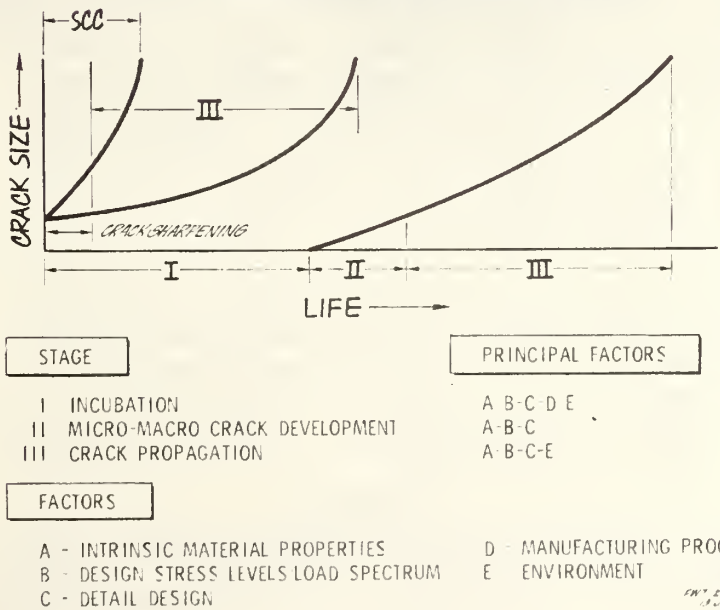
Fig. C-38

THE DESIGNER'S PROBLEM

- ACHIEVEMENT OF ADEQUATE SERVICE LIFE AT ACCEPTABLY LOW RISK IS DEPENDENT ON A LARGE NUMBER OF FACTORS, (some of which are beyond the Direct Control of the Designer) MANY OF THE FACTORS WHICH MUST BE CONSIDERED ARE IN TURN SUBJECT TO A LARGE NUMBER OF VARIABLES. STATISTICAL ASSESSMENTS OF SERVICE LIFE AND RISK CAN BE USED TO ASSIST IN THE DECISION MAKING PROCESS; HOWEVER, WHAT CONSTITUTES ACCEPTABLY LOW RISK HAS NOT BEEN ESTABLISHED.

- TO ACHIEVE DESIRED PERFORMANCE, INCREASING USE IS BEING MADE OF MORE EXOTIC MATERIALS AND MANUFACTURING PROCESSES

    - HIGH STRENGTH METALLIC ALLOYS
    - FILAMENTARY COMPOSITES
    - HIGH FATIGUE RESISTANCE FASTENER SYSTEMS
    - WELDING, ADHESIVE AND DIFFUSION BONDING
    - SURFACE FINISHING

IN ORDER TO ASSURE THAT THE EXPECTED BENEFITS IN SERVICE LIFE ARE ACHIEVED THE DESIGNER MUST HAVE A MORE INTIMATE ASSOCIATION WITH THE MANUFACTURING AND INSPECTION OPERATIONS AND PROVIDE MORE SPECIFIC INFORMATION AND DIRECTIONS.

Fig. C-39

THE DESIGNER'S PROBLEM (Cont'd)

- THE CURRENT TREND TOWARD A DESIGN CRITERIA BASED ON THE ASSUMPTION OF PRE-EXISTING FLAWS AND A DEFINITION THAT FATIGUE LIFE IS BASED ON THE TIME A FLAW BECOMES OF DETECTABLE SIZE REQUIRES NEW KNOWLEDGE ON THE PART OF THE DESIGNER OF THE VARIABLES AFFECTING SERVICE LIFE INCLUDING WHAT THE PRODUCTION MANUFACTURING PROCESSES WILL PRODUCE AND IN- SPECTION CAPABILITIES WILL FIND

- TO AVOID THE PROBLEM OF CATASTROPHIC FAILURE IN MONOLITHIC STRUCTURE THERE HAS BEEN A TREND TOWARD CONSIDERATION OF FAIL SAFE DESIGNS THIS DESIGN CONCEPT HOWEVER PRESENTS A NUMBER OF PROBLEMS
    - PERFORMANCE DEGRADATION (Weight)
    - COST - INITIAL AND MAINTENANCE
      (Additional Piece Parts and Joints)
    - VERIFICATION BY TEST OF FAILURE MODES

A FAIL SAFE STRUCTURE IS NOT FAIL SAFE UNLESS THE FIRST FAILURE CAN BE READILY DETECTED AND CORRECTED

Fig. C-40

SOME COMMENTS ON FAIL-SAFE DESIGN

Walter S. Hyler
Battelle-Columbus

Abstract: Tensile design properties are based on strength
distribution curves with mean stress and standard
deviation. No similar curves are available for
fracture resistance. Data are presented to show the
large scatter of K_{1c}, depending on slight processing
variations as well as on location within the same
sheet. A coefficient of variation indicates whether
processing is well under control. Improved material
characterization will be necessary for fail-safe
design with thick, moderately thick, and thin
structures--again requiring close interrelation
between design and materials and processing.

The talk by Mr. Dietz this morning was very impressive to
me, probably not only because it was done so well, but also
because it touched upon many concerns that I have relative to
fail-safe or damage tolerant design. One thing which came out
clearly was his statement about the need for positive interaction
between the designer and the materials and process engineers,
since in damage-tolerant design of high-performance systems, we
are concerned with fracture and fatigue-crack propagation for
which analytical methods still are being developed and for which
optimum materials processing is necessary. This is where I
wanted to start my discussion and what I wanted to amplify on
because I believe that it really is a fairly critical point.

In the old days--and I say this with a smile because I really
don't know about structural design in the old days--it seemed
we were interested in ultimate strength, in limit load, and in
developing positive margins, and then we would go on to the
next design problem. With our high-performance airplanes, we
have more complexities. We may still be computing ultimate
load, limit load, and positive margins, but with damage tolerance
we are also concerned with establishment of minimum detectable
flaw sizes, crack propagation by fatigue, establishment of
inspection intervals and useful life, and the strength of flawed
structure.

For handling ultimate and yield load design calculations,
we have learned to employ and have become comfortable with the
use of design allowables based on statistically established minimum
design properties. It is obvious that we have not nearly approached
that point yet in terms of fatigue-crack propagation and fracture;
but as we evolve design concepts, such as fail-safe or damage
tolerance, then I think that we will have to give some attention

to the establishment of minimum design allowables for these behaviors.

Both fatigue-crack growth and fracture are sensitive to geometric factors in component geometry, to composition and structure of the material, and to processing details in the fabrication operations. The measures of these behaviors, such as K_c, K_{Ic}, and da/dN versus $\triangle K$ have grown out of the development of fracture mechanics as a stress-analysis tool and can be used in design analysis. In the discussion today, I make reference to only one of these measures, K_{Ic}; however, the philosophical implications also apply to problems associated with plane-stress fracture and fatigue-crack propagation.

While K_{Ic} has historically been considered a constant for a material under a plane-strain stress state, this is an over-simplification when one considers the reality of materials in structural components. If one simply considers a material in a product form, such as plate, processed by procedures reasonably well under control, we know from countless tensile tests that there is a variation in tensile properties of 15 to 40 ksi, depending upon the material and its strength level. Similarly, one can expect the same material to exhibit a variation in K_{Ic} of some appreciable, but generally unknown, magnitude. Now consider the same thick plate material fabricated into a useful but complex piece of structure with distinct changes in thickness and plan-form and finally heat treated to the desired strength level. If the material is quench-rate sensitive or sensitive to some other processing procedure, one can expect that the variability in K_{Ic} from location to location may be appreciably greater than in the prior case. A design philosophy such as damage tolerance that is based in part on fracture toughness considerations is not on too solid a footing unless there is a realistic appraisal of this variation.

Figure D-1 provides some rationale for the above statement. In this figure are shown for three generic materials (with their representative F_{tu} levels), the critical surface crack length associated with various K_{Ic} levels. If one decides, based on limited testing of simplified shapes that "reasonable" K_{Ic} levels should be 60, 60, and 25 ksi inch (for steel, titanium, and aluminum, respectively), and if the real material in a complex component shows variations of $^+_-30$ percent or more (which is not unlikely), the possibility exists that a flaw will grow to instability long before it can be detected with high probability. This is a consequence that may accompany inadequate determination of minimum K_{Ic} values. For example, for the steel alloy in Figure D-1, the carck length is shown to be 0.16 inch. However, if the distribution of K_{Ic} in real parts can vary by $^+_-30$ percent, application of the limit load stress might result in failure with a surface crack of only 0.07 inch; hardly a detectable crack with high probability.

For static properties, as stated before, the aerospace industry has become accustomed to the use of statistically based design allowables, such as the A and B values in MIL-HDBK-5. I would like to review briefly some features of A and B values, since some of the ideas are instructive in considering fracture and fatigue-crack propagation. Figure D-2 shows the analysis of tensile yield strength for 7075-T6 sheet.

In this figure are shown two bell-shaped distribution curves based on an analysis of test data. The solid curve was obtained from test samples sectioned from one sheet. The dashed curve came from a large number of quality control tests on many lots of the alloy. It is seen that in either case the properties fall in a broad range, and for this collection of data the single sheet data covered over half the range obtained from the multilot test data. It is from this type of analysis that the aluminum industry developed the concept of A values as a minimum design mechanical property. Thus, the A value is a minimum strength above which it is expected that 99 percent of all future production will lie with a confidence of 95 percent. I have shown on the figure what I terms an A' value that represents a maximum strength value, below which 99 percent of all future production will lie with a confidence of 95 percent.

For this alloy, the "process capability" is such that 98 percent of all production is expected to lie between A and A'. The term "process capability" implies that for a given production process schedule, there will be a recognizable distribution in strength properties, definable by a reasonable population of test data. The range of properties reflects acceptable limits on composition and structure and on all the controls utilized in each processing step.

When we consider fracture as exemplified by K_{Ic}, it may be that for small quantities of a given material, heat treated carefully and consistently, that K_{Ic} will be a reasonably constant value. If one can convince one's self that such values represent the material in a complex structure, it (K_{Ic}) provides a good engineering tool to employ in damage tolerant design. However, many modern aerospace structural materials are sensitive to various states of processing so that unless one knows the "process capability" of a material with regard to fracture, one may be led to unsafe design conclusions. The next several figures and accompanying discussion amplify this point.

Figure D-3 shows the distribution curve obtained from quality control data on current production Ti-6Al-4V mill-annealed plate. The statistical analysis resulted in the A, A', $\overline{X}$, B and B' values indicated. One concludes from the analysis that the process capability of this alloy is such that 98 percent of expected future values will lie between 118 ksi and 146 ksi.

Figure D-4 shows the relationship between F_{ty} and K_{Ic} for the same titanium alloy in about the same thickness range. In this figure, each horizontal line, vertically tic marked at the ends, marks the minimum and maximum K_{Ic} value observed for a given plate with the appropriate TYS. Two or more tests were conducted for each plate. The excessively large scatter is disturbing since the material is essentially that which can be purchased according to the confines of present military specifications and, as such, corresponds reasonably well in tensile properties to the material shown in Figure D-3. Consequently, the ordinate scale on Figure D-4 is tic marked to indicate A', B' $\overline{X}$, B, and A values from Figure D-3.

From data, as shown in Figure D-4, one must establish a representative K_{Ic} value for use in damage-tolerant design considerations. There are several possibilities for this selection:

(1) Assume a typical value. A natural selection would be a K_{Ic} value associated with the mean strength, $\overline{X}$; thus, a K_{Ic} of about 65 ksi $\sqrt{}$ inch.

(2) Establish a trend line, such as the downward sloping solid line in Figure D-4, and select a minimum K_{Ic} value associated with the A' value; thus, a K_{Ic} of about 42 ksi inch.*

(3) By statistical analysis, using regression techniques and introducing probability and confidence (not necessarily 99, 95), establish a minimum curve and select a minimum K_{Ic} value associated with the A' value. Such an approach would yield a K_{Ic} of 35 ksi $\sqrt{}$ inch, or less.

(4) The fourth alternative is to lump all of the data and compute by statistical techniques the $\overline{X}$ and s associated with the data and a minimum value based on some agreed-upon probability and confidence. A K_{Ic} value about the same as in Alternate 3 would be anticipated.

Neither Alternates 3 nor 4 provide a design value for K_{Ic} that would excite the designer (based on this data collection) since this results at a limit load stress of $F_{tu}/1.5$ in the need to confidently find surface cracks less than about 0.25 inch according to Figure D-1. Alternates 1 and 2 appear less

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* The A' value is chosen since the TYS distribution indicates that this is an expected value for this alloy.

desirable, since they ignore the distinct probability that substantially lower values of K_{Ic} will actually be present in the material (particularly for Alternate 1). There is some belief that for fracture critical components the assumed Alternate 1 value can be screened with quality-control tests. This rationale appears to be potentially erroneous if the distribution of K_{Ic} values in a single plate shows a range of expected values nearly as broad as that from a large sampling of plates -- along the lines suggested for TYS in Figure D-2.

It should be stated that other information than shown in Figures D-3 and D-4 (relative to composition and microstructure and their influence on K_{Ic} for this alloy) suggests that the extensive scatter may not be real for this alloy, providing some further examination of process control is done in order to sort out what contributes to the variability and how process controls can be modified to lessen variability. This is the essential point that Mr. Dietz was addressing in promoting much closer interaction between design people and materials and processing people.

Another material of considerable interest these days is D6AC, a modification of 4340. Figure D-5 shows the distribution curve for yield strength for this alloy based on a large sampling of producer and user quality control data. It was a normal distribution with $\overline{X}$ at 214 ksi. The process capability was such that 98 percent of future production would be expected to fall within the range 200 ksi to 228 ksi with a confidence of 95 percent.

Figure D-6 begins to delineate the real-life situation for this alloy with regard to fracture, or K_{Ic}. This is a very busy figure that requires some preliminary orientation. The rectangular border represents the boundary of 16 different plates of D6AC (2 feet by 3 feet) all processed similarly by General Dynamics/Fort Worth and identically heat treated according to a procedure that I have identified as heat treatment A. Each plate had been laid out to provide a variety of specimens for tensile tests, fracture toughness, etc. On Figure D-6, each of the sixteen plates is identified by a distinct symbol. The symbols on the drawing represent the approximate location in each plate from which a fracture specimen was taken (usually more than one fracture test was conducted and Figure D-6 contains all of the data). The numerical values adjacent to each symbol represent the observed K_{Ic} values.

Within the circle at the lower right-hand corner of the figure are data for six different plates. It is seen that K_{Ic} for the plates at this confined location ranges from about 59 to 86 ksi $\sqrt{inch}$. Within the circle at the lower left-hand corner also are data for six plates, where some plates in this location duplicate those in the right circle. For this second circle,

the range in K_{Ic} is from about 55 to 91 ksi $\sqrt{inch}$. Where plates are represented in both circles, a plate with K_{Ic} on the high side of the range in the right circle may not be similarly ordered in the left circle. In fact, if one were to draw a number of circles at various other locations on the plate plan-form and examine the details of the data within each circle, comparing the data with that in other circles, one would conclude that with this particular heat treatment a fairly large variation in K_{Ic} can be expected within a plate and among the many plates. Detailed perusal of the figure shows this range to be from slightly less than 50 ksi inch to slightly more than 90 ksi inch.

Figure D-7 shows the same fracture-toughness data plotted as a function of TYS, with the A', $\overline{X}$, and A values from Figure D-5 tic marked on the ordinate. Of the 16 plates, 11 had accompanying tensile test data, so that the horizontal lines represent the plate tensile yield strength; the tic marks at the ends are the maximum and minimum K_{Ic} values in the plates from two or more tests. The lowest horizontal line on the figure is for the five other plates, where TYS is an assumed, but reasonable, value. It is evident from this display that the heat-treat process capability with regard to TYS is quite satisfactory. However, one concludes that although the fracture toughness information also is consistent, the inordinate scatter in results may prejudice the utilization of the material since the capability of heat-treatment A would yield a design allowable for K_{Ic} of approximately 37 ksi inch (for 99, 95 on probability and confidence).

Figures D-8 and D-9 illustrate a happier story. Figure D-8 shows K_{Ic} data from eight plates of D6AC that were identically heat treated, but in a different way than the previous 16 plates. This heat treatment, I have identified as heat-treatment B. The plotting scheme in Figure D-8 is the same as for Figure D-6. Examination of the somewhat fewer data in Figure D-8 shows (1) an encouraging uniformity in fracture toughness within a plate and among the plates, and (2) a substantial increase in fracture toughness. The range in K_{Ic} values is from about 91 to 102 ksi $\sqrt{inch}$.

Figure D-9 shows the K_{Ic} values plotted against TYS as in Figure 7; however, there were tensile data for only two of the plates. The middle line at TYS of about 220 ksi is for the remaining six plates. Since so few tensile data were available, it is difficult to say any more than that there appears to be a greater spread in TYS associated with heat-treatment B than for A. More important, however, while the fracture test results for D6AC with heat-treatment A suggest a possibly severe sensitivy to nonuniform quench rates along the plan-form and from plate to plate, heat-treatment B suggests that its process capability is compatible with achieving high and uniform fracture toughness.

Not only that, but a designer familiar with the use of statistical design allowables for static strength would have a much more comfortable regard for an "A" value for fracture toughness of about 83 ksi $\sqrt{inch}$ to which manipulation of these data lead one.

This type of materials and processes research, which has been conducted at General Dynamics/Fort Worth, interfacing with the stress and structures people is the way that confidence will be built into the use of the damage tolerant design concept.

Figure D-10 is illustrated to recap some of the statistical measures that were employed in preparing several of the other figures. Information shown for the three materials and two strength attributes are coefficient of variation, C; average strength, $\bar{X}$; and standard deviation, s. As one examines the right-hand column (s), it is immediately evident that the standard deviation for fracture toughness of heat-treatment A for D6AC is excessively high, certainly suggesting that processing is yielding an intolerable variability in results. On the other hand, heat-treatment B exhibits an s value even less than that for TYS, when processing has been examined and developed for the strength attribute of importance.

In the column for coefficient of variation, which is a measure of dispersion, s, normalized by the average value of strength, several features stand out. First, the C value for Ti-6Al-4V alloy for TYS is in excess of 50 percent higher than the comparable value for the other two materials. This suggests, based on this and other information, that process control for this alloy probably can be improved. This process control, in turn, may significantly alter the fracture toughness display of Figure D-4. Recent work at North American Rockwell relative to the B-1 is beginning to bear this out. Finally, the almost order of magnitude difference between C values for TYS and fracture toughness (A heat treatment) for D6AC reinforces the idea that process development and control for fracture toughness was needed.

This discussion has been an attempt to quantify, by examples, my thoughts and those expressed independently by Mr. Dietz and others at this meeting that the development of the damage tolerant design concept presents a challenge in the aerospace industry that encompasses design engineers as well as materials and processing personnel. The challenge is not only in developing and understanding the design concepts utilizing fracture mechanics. Equally important is to recognize that the mechanical behaviors associated with damage tolerance, fracture toughness, and fatigue-crack propagation are structure and process sensitive. This sensitivity will require perhaps an order of magnitude increase in materials and process research, development, and characterization on aerospace systems contracts in order to assure that the material in future structural components will have high and uniform fracture toughness.

FIGURE D-1

CRITICAL SURFACE-CRACK LENGTH VALUES FOR SEVERAL GENERIC MATERIALS WITH ASSUMED K_{IC} VALUES

Steel, $F_{tu} = 280$ Ksi		Ti, $F_{tu} = 130$ Ksi		Al, $F_{tu} = 75$ Ksi	
K_{IC}	2C, inch	K_{IC}	2C, inch	K_{IC}	2C, inch
20	0.017	20	0.078	15	0.14
40	0.07	40	0.31	20	0.25
60	0.16	60	0.70	25	0.39
80	0.28	80	1.24	30	0.56
100	0.45	100	1.70	35	0.77

Based on semicircular surface flaw and $S = F_{tu}/1.5$ and $K_{IC} = \dfrac{S}{1.57}\sqrt{\pi C}$
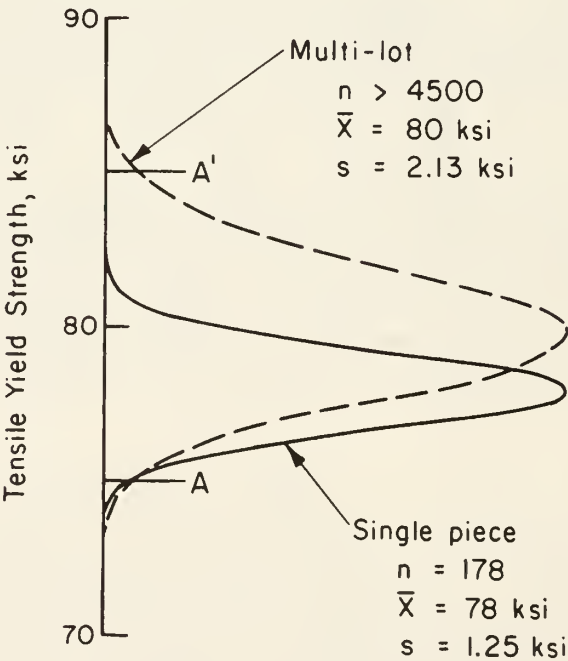
ALCLAD 7075-T6 Sheet

FIGURE D-2. DISTRIBUTION OF TENSILE YIELD STRENGTH VALUES FOR ALCLAD 7075-T6 SHEET
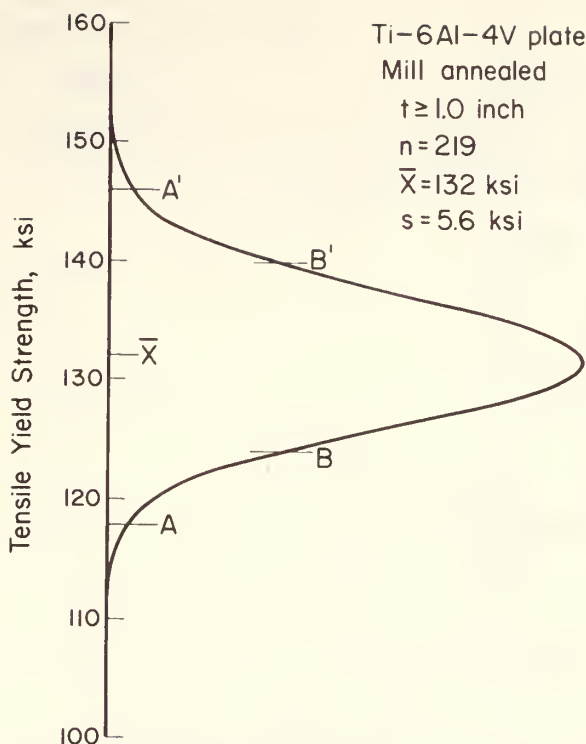
FIGURE D-3. DISTRIBUTION OF TENSILE YIELD
STRENGTH VALUES FOR Ti-6Al-4V
PLATE

K_{Ic}, Plane Strain Fracture Toughness
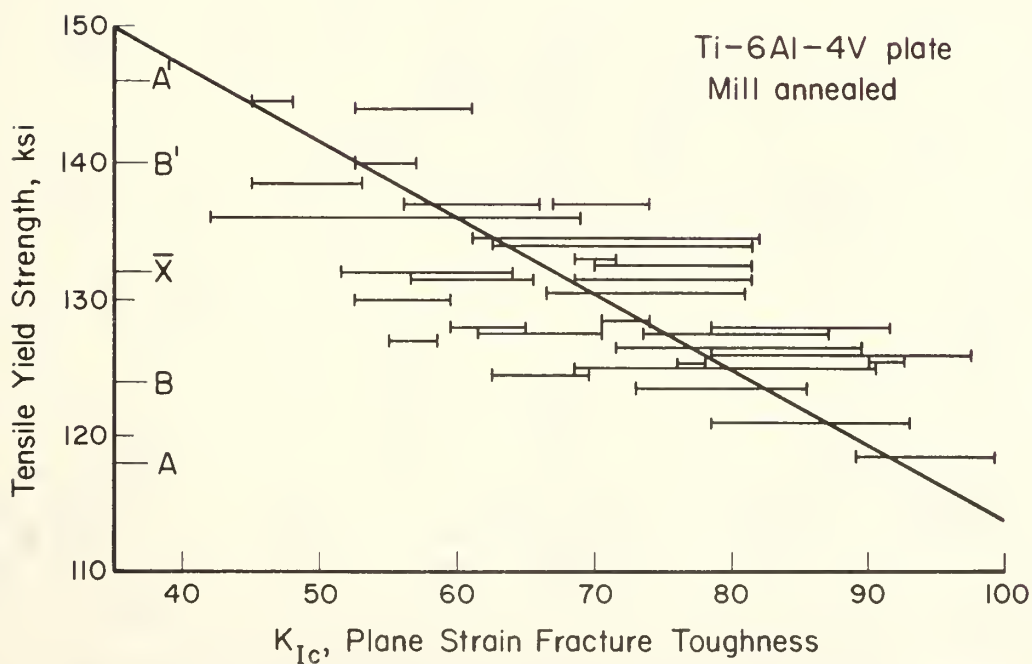
FIGURE D-4. RELATIONSHIP BETWEEN TENSILE YIELD STRENGTH
AND K_{Ic} FOR Ti-6Al-4V ANNEALED PLATE

D6AC plate
(Nominally 220–240, ksi)
n=3684
$\overline{X}$=214 ksi
s=5.73 lksi
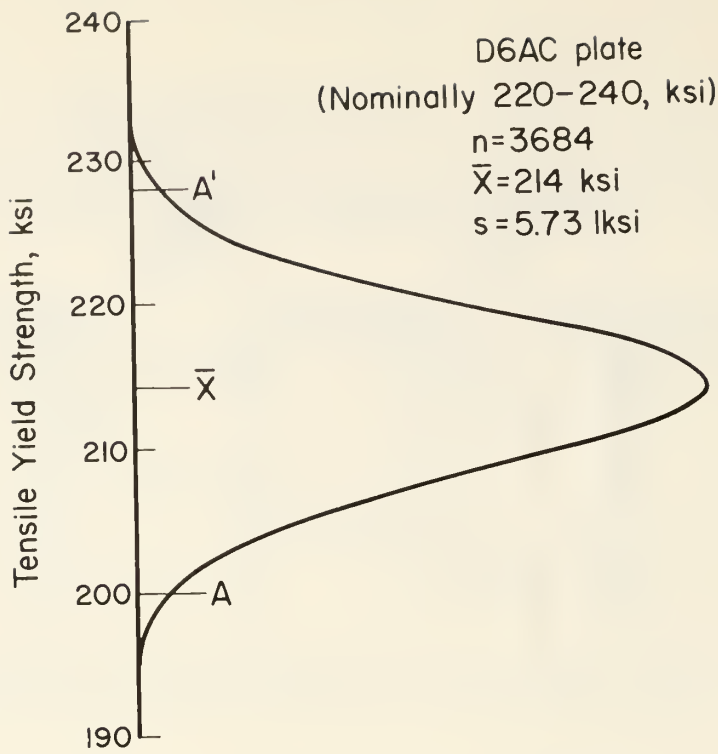
FIGURE D-5. DISTRIBUTION OF TENSILE YIELD
STRENGTH VALUES FOR D6AC
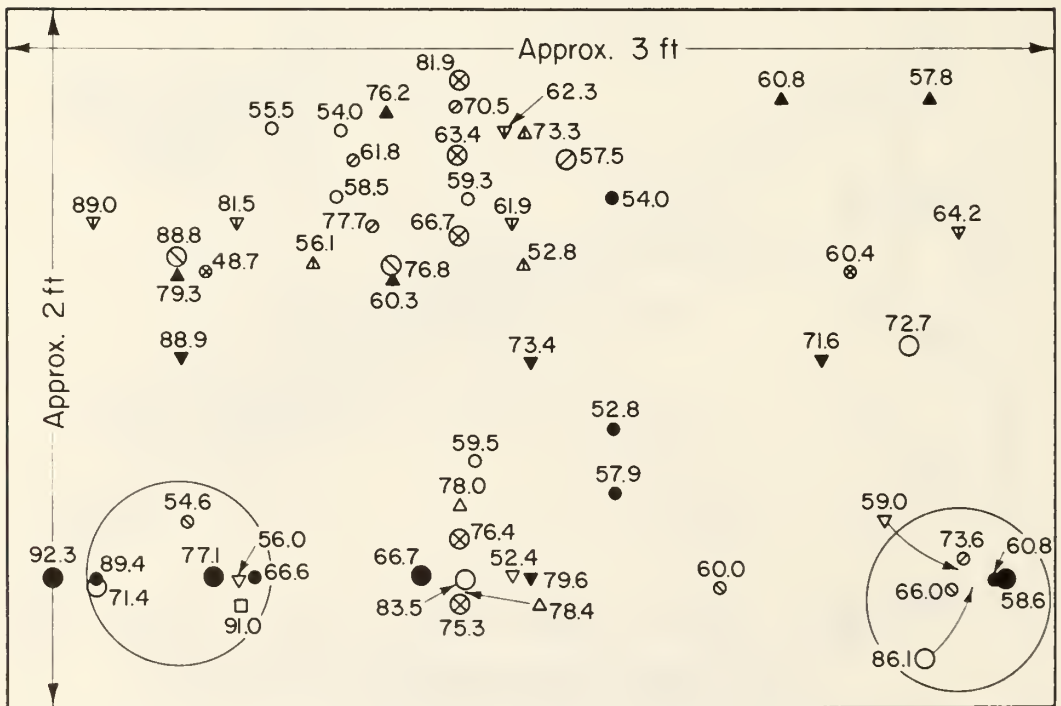
FRACTURE TOUGHNESS DISTRIBUTION IN SIXTEEN PLATES OF D6AC
FIGURE D-6
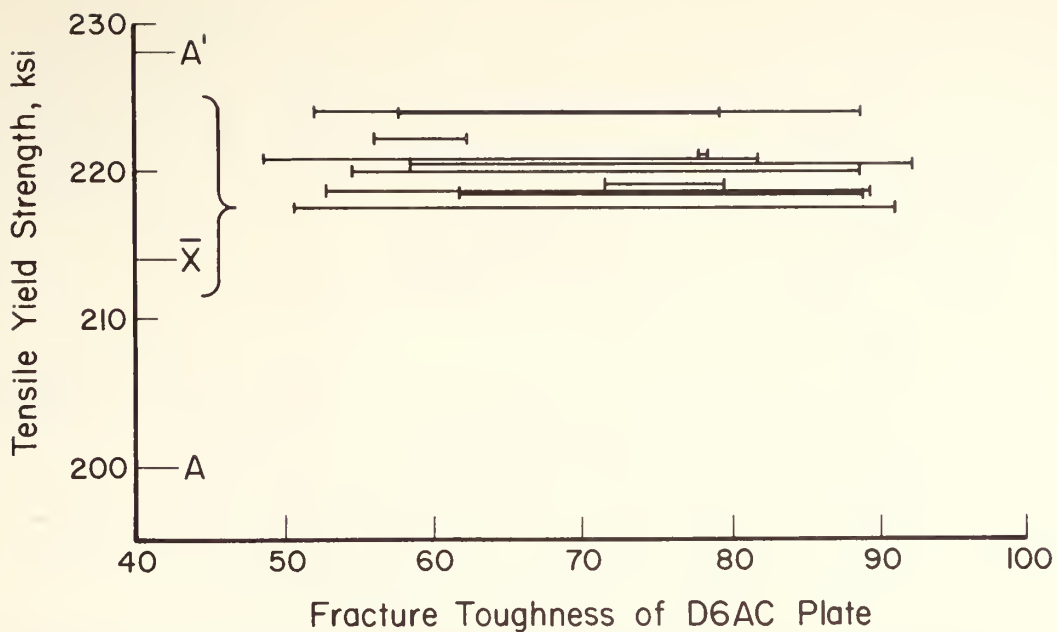
FIGURE D-7. RELATIONSHIP BETWEEN TENSILE YIELD
STRENGTH AND FRACTURE TOUGHNESS
OF D6AC PLATE, HEAT TREATMENT A
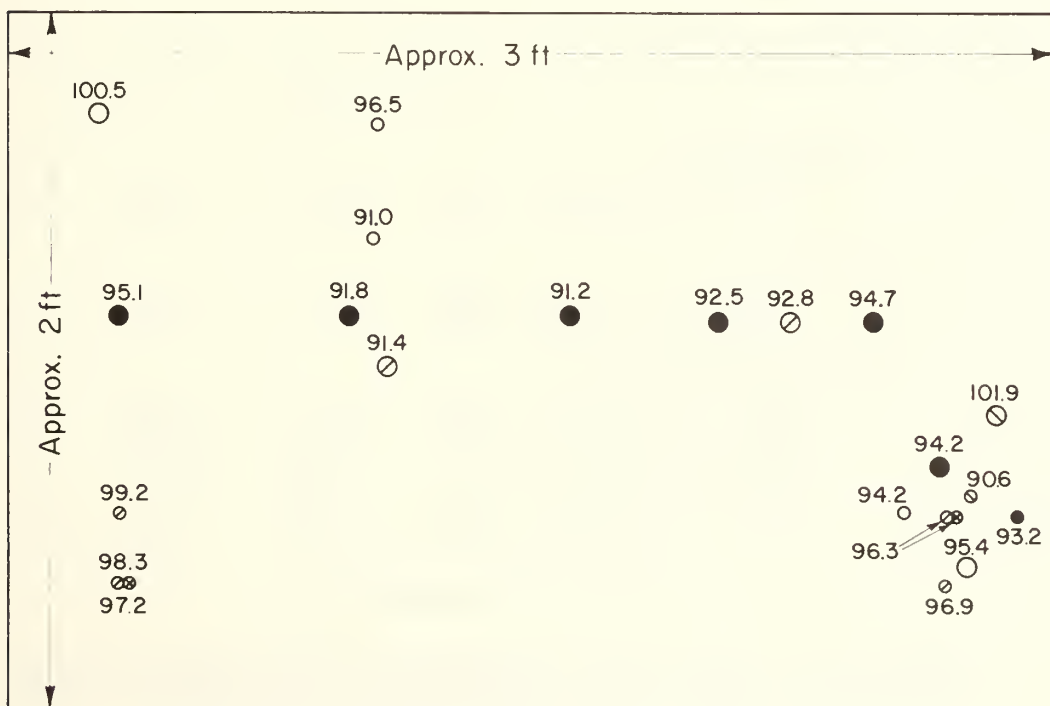
FRACTURE TOUGHNESS IN EIGHT PLATES OF D6AC

FIGURE D-8
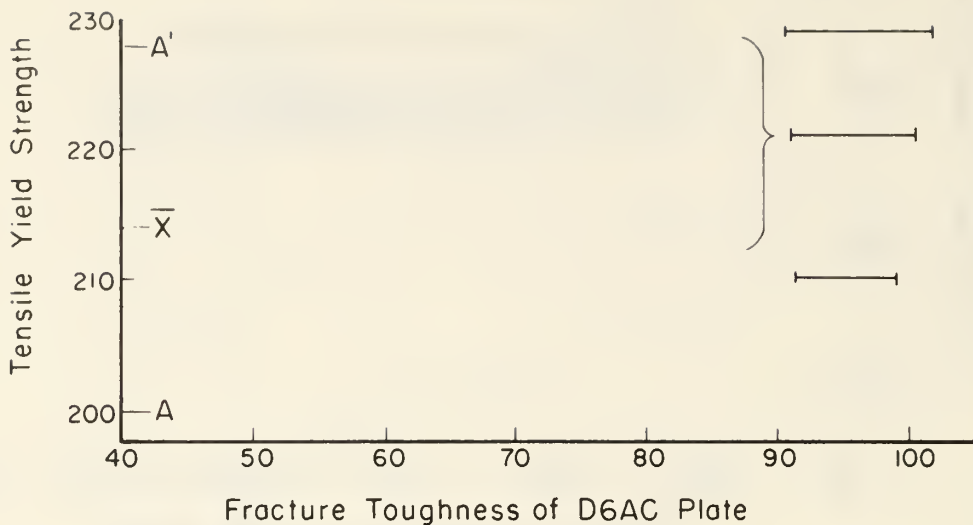
FIGURE D-9. RELATIONSHIP BETWEEN TENSILE YIELD STRENGTH AND K_{Ic} FOR D6AC PLATE, HEAT TREATMENT B

	C*	$\overline{X}$	s
ALCAD 7075-T6			
Tensile Yield Strength —	0.027	80	2.13
Ti6AI-4V			
Tensile Yield Strength —	0.042	132	5.6
D6AC Steel (220-240 Ksi)			
Tensile Yield Strength —	0.027	214	5.73
Fracture Toughness, A —	0.17	64.5	11.0
Fracture Toughness, B —	0.042	94.8	4.0

$$C = \text{Coefficient of Variation} = \frac{s}{\overline{X}}$$

FIGURE D-10. STATISTICAL MEASURES FOR THE THREE ALLOYS

IMPACT OF PROCUREMENT PRACTICES
ON C-5 STRUCTURAL DESIGN

W. T. Shuler
Lockheed-Georgia

Abstract: Contractual factors which affected the structural
design of the C-5 are analyzed and evaluated, including
guaranties on performance, low weight empty, and crack-
free fatigue life as well as fixed price contract and
program for concurrent development and production. No
revolutionary change in procurement practices is recom-
mended, but areas of possible improvement are indicated
and discussed.

Only the abstract is available.

STRUCTURAL DESIGN CONSIDERATIONS FOR
A SPACE SHUTTLE

F. F. W. Krohn
General Dynamics - San Diego

Abstract: Details of structural arrangement and thermal protective
system for a space shuttle booster are shown and
corresponding design conditions are given including
thermal expansion, creep, shock impingement, acoustic
environment. Design considerations for high-temperature
materials, and structural features for components
incorporating a safe-life or a fail-safe approach are
discussed -- indicating a new order of complexity
compared to aircraft design.

I would like to present to you design considerations for a
space shuttle booster. The design requirements on space shuttle
vehicles are a unique combination of the design requirements for
aircraft and rocket launch vehicles. This is something brand new
and the combination of these requirements will impose on us quite
a number of new problems in materials and design.

For those who might not have followed the space shuttle
project, there have been four teams engaged in design studies.
McDonnell-Douglas has a design effort on the booster as well as
on the orbiter in Phase B. North-American Rockwell and General
Dynamics Convair have a split in tasks, with us at General
Dynamics doing the booster and North-American doing the orbiter.
Then there is a team of Grumman and Boeing and the fourth is one
company again -- Lockheed.

The objective of the space shuttle system is to put payload
cheaply in orbit with re-usable vehicles. The different versions
which have been studied accomplish this with marked difference
in cost. Some of the systems throw away tanks which weigh up
to 60,000#. Payload cost in orbit varies from $75 for fully
re-usable to $200 plus for partially re-usable systems.

I will address myself to a system which mounts the orbiter
parallel to the axis of the booster -- a polar orbit design
mission with a payload of 40,000# and an orbiter with two engines
requiring a staging velocity of 10,800 ft/sec. This results in
the most difficult booster and this vehicle system was studied
in depth. However, we had a re-direction as we found out rather
lately, through side studies, that it would be much more advantageous
to have three engines on the orbiter. This brings the staging
velocity down to less than 8000 ft/sec, even for the polar missions.
As stated I will, however, address myself to the particular system
with 10,800 ft/sec staging velocity.

The approximate lift-off weight on the pad, with the orbiter mounted, is slightly in excess of 5 million pounds for the combination -- a rather staggering weight. As far as the booster is concerned, our lift-off weight is 4,188,000#. The landing weight is 638,000#. The rest is propellants.

The vehicle is really a huge tanker with 12 rocket engines mounted at its base. To give you some idea about the dimensions, the booster has a wing span of 144 feet and a total length of 256 feet. For cruise back it has JP propellants of approximately 144,000#. It has an ascent propellant weight of 3,382,000#. Of the total vehicle weight, essentially more than half is oxidizer.

Now let us get a little bit into the description of this vehicle system. It takes off vertically. The orbiter is connected with a reversible link system which is sized in such a fashion that the orbiter can be ejected even with the orbiter engines dead. This is accomplished by the thrust on the booster. In normal operation the orbiter has its thrust run up to approximately 50% at time of separation. There is still a very large force in the horizontal direction on the linkage system.

After the orbiter has been separated, the vehicle will assume an angle of attack of approximately 60° and spend its kinetic energy very rapidly. The whole heating cycle is exceedingly short and the heat-up is very steep. Here, you will see, we have very severe implications for structural design. Everything on the outside will be hot and any inside structure will just lag in temperature. The vehicle, during the time of re-entry, is controlled by a set of reaction controls because the tail is fully ineffective and also the wings are not very effective at this time.

We have 12 jet engines in this vehicle. They are deployed and run up one after the other following re-entry. The last engine will have run up approximately two minutes after the first has been deployed. The vehicle then will cruise home or it will cruise to the next airfield. The vehicle has the capability just like some of the orbiters to ferry back to the launch pad.

In structures arrangement there is a very large difference between the design requirements for 10,800 ft/sec staging velocity and a vehicle staging with less than 8000 ft/sec. A 10,800 ft/sec vehicle requires a substantial thermal protection system (TPS). An 8000 ft/sec vehicle can be built around what we call the heat sink principle. Here temperatures are controlled by allocation of sufficient mass in the surface. Use of heavy aluminum gauges will reduce temperatures to 300°F over most of the vehicle surface.

Now let us look at the structural arrangement shown in Fig. F-1. The vehicle has a thermally protected backbone structure consisting of thrust structure, LH_2 tank, intertank, LO_2 tank and forward tank extension. All major vehicle loads are introduced into this thermally protected core.

The wing is link-attached with vertical and drag links. Once we fuel the LH_2 tank, the tank wall will go to approximately -100°F and shrink very substantially while everything around it will stay at room temperature. So, all supports have to be statically determinate.

The orbiter attachments are backed up by two large bulkheads which are essentially external in the region of the LH_2 tank and one very deep forward bulkhead which is inside the LO_2 tank. This forward support takes the brunt of the loading.

Figure F-2 shows the heat shield which shrouds the body. It is composed of large semistructural elements. We did not follow the usual route of having TPS panels of 2 or 4 feet square. Instead we have large panel shells with slip joints and a horseshoe element directly attached to the wing because the temperatures of the upper wing surface and this shroud are approximately the same. One must make sure to get compatible expansion. The best way to enforce this is to attach the heat shield to the wing directly as a root web. All other shell elements are supported in the center and expand fore and aft with respect to the substructure.

The great advantage of this arrangement is that it reduces the potential leak area in seals and slip joints to less than 1000 ft length compared to some miles in a conventional panel system. This makes it possible to control the containment of purge gas. There is a purge envelope over the LH_2 tank separated by a bulkhead from the purge cavity over the LO2 tank. Both cavities are filled on the launch pad with dry nitrogen.

In up-flight, we bleed off the purge gas to avoid large internal pressures on our heat shield. Just very shortly before staging, the bleed-off vents are closed and kept closed during the time of maximum heating. We will have some loss of purge gas during this time without any doubt. We have a self-imposed spec of permissible leakage area (not to exceed 350 square inches) and this permits us to go through this environment with a quite high degree of assurance that we will not get hot plasma into the purge cavities. After we have suffered through the re-entry, the tank pressure will be reduced and the vents will be opened to take air on board for pressure equalization. So much for the general description and operational aspects of the vehicle effecting the structure.

The design criteria shown in Fig. F-3 indicate for the launch mode an ultimate factor of 1.4 x limit and for the aircraft mode an ultimate factor of 1.5 x limit. Buried in this arbitary limit-to-ultimate factor are 80 million dollars. That is the cost to conduct a full-scale separate static test.

Now, what are we really after? We are after a vehicle which is capable to take limit loads at the end of its useful structural life. We have proposed to the contracting agency to leave these factors in for design but to drop the whole static test program except for development testing. Instead, we would go through four lifetimes of operational flights and then subject the structure to 1.15 times limit load. We feel that this is a much more useful type of testing. Arbitrary design criteria which we have set up with the ratio of yield-to-ultimate factor on one outdated aluminum alloy in mind most certainly are suspect in our time and age.

The tanks, which compose a very large portion of the structure, are not really designed by these external load factors. They are designed by fracture mechanics or, to put it another way, by proof-testing requirements. The tests are conducted with a proof pressure which precludes the presence of a crack which could grow to critical size with actual operational pressures and projected crack growth during 150 flights. This imposes a very severe burden on the materials community because now we really need to know crack growth rates and cannot be content anymore with K_{1c} alone. All our gages are thin enough to be neither plane strain nor plane stress, but something in between. And here we have a very substantial lack of data, even on the 2219 material which we have selected and which is excellent for the application.

One of the new criteria coming in is creep. This is a wide open field. We have limited our creep to .2% plastic deformation. We do not know if this is right because we don't know what really happens with a panel that has experienced this much creep. One could have panel flutter. One also has steadily changing conditions as far as overall dynamic behavior of the structure in flutter is concerned. We have designed around many of these problems, just to be on the safe side.

Figure F-4 shows peak limit load intensities and the load envelope of the vehicle with 25 different loading conditions. As you can see, not much is designed by the aircraft mode of operation, mostly localized structure only. Figure F-5 shows that the tanks are designed for safe life and only a relatively small portion of the structure is designed for fail-safe criteria.

We made a valiant attempt to get fail-safe tanks and looked into crack stoppers as they are presently employed on commercial jet fuselages. We found that they would have to be at a prohibitively

close spacing of approximately $4\frac{1}{2}$". We were quite uncomfortable, as aircraft designers always are, with all this safe life situation.

Then, by fracture mechanics analysis, we found to our delight that the tanks will leak long before we get to critical crack size. We really could fly with a subcritical, leaking crack on quite a number of missions before it would become critical. We have to make very sure that we install sensors in the purge cavities to find out if we have somewhere a slow leakage. This should warn us and we then could take a closer look at the tank wall.

Figure F-6 shows the structural backbone of the vehicle with thrust structure, LH_2 tank, intertank, LO_2 tank, and forward tank extension. Orbiter support points as well as landing gear attachments are shown. A heat shield is mounted approximately six feet away from the rocket nozzle exit plane. Before we go into these components and their design logic, let us have a look at the crew compartment.

Figure F-7 shows the structure of the crew compartment which, however, is not designed to structures requirements at all for a very unique reason. Severe noise levels caused by the captured shocks between booster and orbiter noses require this component to be designed for noise suppressions, e.g. with heavy skins.

The crew compartment and the electronic compartment behind it are suspended from the outer heat shield shell which will expand substantially during heat-up in re-entry. The shell is conventional aluminum alloy construction. We have fused silica glass windows and there is a local area around them which is designed to thermo-structural requirements. The side enclosures can be swung open and the crew can eject out to both sides. This is required since the orbiter is mounted on top. Seats can be swiveled for vertical and horizontal flight.

We have stringers, very few of them, on the inside and the frames are on the outside. The whole enclosure is enveloped with low-density fiberglas insulation and 1" dynaflex over it to maintain the temperature environment in the compartment.

Figure F-8 shows the forward tank extension with nose gear support and a large JP tank located here for stability during re-entry. The forward segment of the heat shield is supported here with two links and guide rollers. When we re-enter, the heat shield which is bolted to the sub-structure at the rear end of the lox tank, expands at this point approximately 6" forward. Tank extension construction material in the forward portion is 2024. In the aft portion it is 2219, principally for compatibility with the wall of the lox tank which will see temperatures of -320°F.

Figure F-9 shows the lox tank which is made of 2219 aluminum and designed principally for fracture mechanics considerations. The nose gear loads do not tax this structure very much at all and there is a minimum amount of stiffening. We have integral T stiffeners longitudinally and truss-type frames. The reason for not going to a web-type frame is, of course, that the tank shrinks and we had to tailor the stiffness of the frame very carefully in order to prevent large pull forces on the tank stiffeners in chill down. The rather massive bulkhead at the orbiter forward attachment was mentioned before.

All fittings are designed of two elements. Why not three for fail-safe? The reason is that we always would have covered one of the elements and therefore could not have inspected one of the elements visually.

The lateral load on the forward orbiter attachment is 1.25 million pounds limit and also the vertical load is very substatial but, nevertheless, all this does not design the structure here because stiffness requirements are critical. We had to meet a natural frequency with the orbiter mounted on top of approximately .8 cps. We made .9 cps after many trials and many computer runs.

No large thermal stresses are encountered on a lox tank because the whole structure system will cool down simultaneously. We are definitely working in the mixed mode as far as fracture mechanics is concerned. Tank walls are approximately 1/8" in the upper portion. There are also quite a number of baffles mounted in the tank.

Figure F-10 shows the intertank section. It connects the LO_2 tank with the LH_2 tank. It is crammed full with equipment. Again we are dealing with an aluminum structure, local reinforcements in titanium, and here, fortunately, we can go inward and get our stiffness by depth. So, no advanced high stiffness materials are employed.

The canard and the orbiter drag link are attached in this region. The all-movable canard is streamlined during re-entry since reaction controls are used. This effectively eliminates temperature differences between upper and lower surfaces. Regarding the critical orbiter connections we are becoming quite cautious and are toying with the idea to subject any of the critical orbiter connecting elements to proof loads for qualification.

Figure F-11 shows the LH_2 tank, made of 2219 aluminum, approximately 120 feet long and 33 feet in diameter. On the outside of the tank we have the frames, including the orbiter aft attachment bulkheads.

These two bulkheads experience their main load at the moment of separation but, interestingly enough, the exceedingly turbulent flow around these vehicles forced us into the ultimate of stiffness design on the frames because otherwise we could probably have fatigued this area half-way through the first flight at the point where the stiffeners join the frames. Whereas we quite succeeded in meeting the overall natural frequency requirements for the combined booster and orbiter with aluminum frames in this location here, we now had to beef up the caps with beryllium for local stiffness.

Of course, we could take other materials here. It is beryllium because its price has come down substantially and we laminated it. Even if we should have a crack in one of the laminations we still do not lose our stiffness because a very localized area would be effected. Of course, we could have taken another route by lowering the height of the stiffeners joining the bulkhead here but this would have resulted in a substantial weight penalty.

Figure F-12 shows the landing gear support structure. Essentially it is a wheel well, structurally fully enclosed. Vertical and drag links introducing the loads into bulkheads and into longerons which are integral with the tank wall provide an attachment system which permits the decrease in dimensions which takes place when we fuel the tank.

But principally we selected this system because of the need for a meaningful drop test arrangement without involving too much of the vehicle. The arrangement permits drop testing the landing gear separately and with a concrete barrel on top of it because it is supported in a determinate fashion and loads into the structure can be determined accurately. The cost of testing requires very careful consideration and is quite important for the structural design of future large vehicles.

Figure F-13 shows the wing-body-TPS arrangement. A drag brace taking the drag loads into the longeron of the tank walls is the only fore and aft restraint of the wing and the horseshoe heat shield attaches directly to the root rib here. Wing and heat shield expand from this attachment point. All the dimensions are steadily changing and there are slip joints to arrive at an expansion-compatible structures system.

A metallic seal at the fore and aft end of the TPS horseshoe section is provided by a bellows. We have corrugated skins all over this vehicle because aerodynamic smoothness is of little concern. As a matter of fact, if one would start with a smooth surface on a high-temperature vehicle subjected to flash heating, after the first flight you would not have it any more. This necessary approach makes aircraft aerodynamicists shudder.

Figure F-14 shows the cryogenic insulation system. Again, it is something rather unique. All the cryogenic insulation systems of the past have been one-shot systems but here we need long-time reliability because maintenance in a tank,which is 10 stories high and has 33 feet diameter,could be prohibitive.

An external foam insulation system would not work because its specific heat is such that it would melt due to the high outside temperatures, in spite of the short heating times. So we need the heat sink of the structure to more or less protect the foam. We have a gas layer system without an inner liner, reconciling ourselves from the beginning with the fact that even with a plastic liner hydrogen would penetrate sooner or later during a 10-year life span.

The unique polyphenyene oxide foam has a capillary system into which the hydrogen penetrates. A gas layer is formed which is kept in equilibrium by capillary forces at the surface. The system is structurally beautiful because it does not impose large shear forces on the adhesive joints. This is very important because it is a tough proposition to hold on a cryogenic insulation system at -421°F and we have more than a third of an acre of it. Besides, all the small columns of the system are so nicely stabilized against each other that a heavy man can walk on it.

Figure F-15 shows the thrust structure. This is a fail-safe system with multiple load paths, partly with trusses, partly with shear webs providing a benign mode of failure which a truss does not give us. The basic structure is titanium, reinforced by boron-aluminum. Their coefficients of thermal expansion are close, 5.0×10^{-6} and 3.1×10^{-6}, respectively. Graphite epoxy with a coefficient of 1×10^{-6} is not compatible with a metallic structure and we would have to make the whole article of graphite epoxy.

The problem is also that boron epoxy and graphite epoxy are deficient in high-temperature applications. If graphite epoxy is heated to 350°F in the presence of moisture its strength will deteriorate. We have tested boron aluminum rather successfully up to 700°F. There is an additional consideration. You might have high temperature only for a short time but you need a certain heat sink capability to prevent the temperature from going up too fast. Lately we have done exceedingly well with boron and graphite polyimides which would have a substantial cost advantage but we were not quite sure of them at the time when we conducted this study.

Let me point out the importance of weight savings. For every pound of advanced composite we save one pound of weight in stiffness critical areas. We attempt to put one pound of weight in orbit for approximately $75, say $100. Let us look at the orbiter first. Without any change in sizing, with 100 flights at $100 a pound we generate a value of $10,000. Assuming an average 60% payload factor, this makes it $6000 per pound or, to show definitely a profit, say $5000 per pound.

For this particular booster we need a 5 pounds saving to put one more pound of payload into the orbiter. This means for the booster one pound of weight saved is approximately $1000. It shows that advanced materials at a cost of about $350 per pound don't have to frighten our community as long as we apply them in such an economical fashion. The installed cost of aluminum is anywhere between $35 to 150 per pound, depending on the quantity of production. As we are progressing into this new world of advanced materials, we will probably employ them in more complex shapes. Right now, the easiest way to employ them economically is just as reinforcements in localized areas. I am afraid we will always have a mix of many materials. The world is not built out of one material.

Figure F-16 shows the base heat shield which is very lightly loaded. Here we are really going all-out with exotic materials. There are beryllium truss systems, providing a quadruple load path and in a conservative fail-safe approach. The performance of these trusses on tests was just fantastic and columns with an L/ρ approaching 300 are somewhat of an eerie sight. We have them ball-jointed to prevent lateral loading which shows what you can do if you watch the peculiarities of a material.

The outer region of the bulkhead consists of corrugated rené 41 panels while the center panels consist of coated corrugated columbium. The reason is that the temperatures at the periphery are reduced due to air circulation, particularly on the launch pad. These are smaller panels, different from the heat shield arrangement on the rest of the vehicle. It is a slip-jointed system.

Figure F-17 shows the wing arrangement, housing the engines with cutouts over a large portion of the structure. The acoustic environment of the engines with an output of around 172 decibels requires high stiffness surface panel construction in this area which merits every type of scrutiny we could put on it. The typical wing structure appears to be a throwback to old days with heavy concentrated load members of titanium and semi-structural covers. We have a fail-safe system of ribs and spars and corrugated surfaces with the corragation running in the fore and aft direction.

The reason for this arrangement becomes apparent when we look at the wing structure in Fig. F-18. The skin of Inconel 718 goes up to $1350^\circ F$ and can expand and flex between the attachments at the node points while the heavy titanium spar cap has a maximum temperature of $300^\circ F$. We have to get used to thinking in terms of compatible heat-up because a panel could quickly deform to an unacceptable level and move into a critical panel flutter mode.

Figure F-19 shows the vertical stabilizer which consists of conventional, integrally stiffened type of titanium construction. The thermal environment is rather mild because the vertical surface is shaded during re-entry. Only the leading edge is heat-sinked to cope with plume impingement from the orbiter.

Figure F-20 shows the canard structure which again is rather conventional. The internal structure and substructure skin are titanium but skin gages and outer surface material are determined by thermal environment considerations. This vehicle flies by wire so there are no linkages to the crew department.

Figure F-21 summarizes the design criteria for the booster TPS. However, the creep factor has been lowered in the meantime to 2 life cycles with concurrence of NASA because the penalty would have been very severe and all structures in the TPS are, after all, but secondary types of structure. With a cumulative creep of .2% and a factor of 2, our plastic deformation in actual flight experience will be .1%. Panel flutter might become critical after such plastic deformation. Flight monitoring and replacement, if required, should be considered since actual conditions cannot be simulated in laboratory type tests.

Figure F-22 shows the pressures to which the outer TPS is subjected. They are quite substantial on the lower surface but rather mild on the upper surface.

The heating environment is shown in Figure F-23 and you can see that we are only approximately 100 seconds above 1000°F, that's all. So you heat up and you cool down and everything on the inside will lag substantially. Consequently, the influence of the local heat sink on or near the outer surface will assume a remarkable influence on the temperatures which we actually experience. Figure F-24 shows a plot of TPS temperature versus thickness for corrugated and smooth TPS. With increasing gage thickness the temperature goes down so you really can tailor your temperatures for a vehicle like this.

Now let us look to what we have tailored these temperatures. Figure F-25 shows the booster body temperatures for the TPS. We can use rené 41 throughout and heat sink it down to the criticality of this material. The required maximum gage we have determined is .058. On the wing, the situation is a bit different and rather serious due to nose shock impingement. Temperatures and various materials which have been used on the aerodynamic surfaces are shown in Fig. F-26.

Figure F-27 shows what this flash heating will do to various types of TPS panels. This is very interesting. The total stress consists of thermal stresses due to temperature gradient and bending stresses due to air loads. For the first configuration

of .016 rené 41 the total stress is 43,700 psi compared with an allowable of 4100 psi. So we applied heat sinking and went to .055 rené 41 which gave us a total stress of 37,700 psi compared with an allowable of 13,700 psi. We still could not make the grade.

We found out that in flash-heating one cannot employ the type of structure which we have used since the beginning of metal aircraft design. So we went to the third configuration shown in Fig. F-27. Because of the reduced moment of inertia it forced us to a much closer frame spacing on the lower surface but with essentially everything working at the same temperature we had a total stress of 11,800 psi compared with an allowable of 13,700 psi.

Figure F-28 shows the frames supporting the body TPS which are about 9" deep. We first had a titanium frame with a corrugated web but after we had found out the very high significance of heatsinking, we went to a quite conventional aluminum construction with additional beryllium caps to maintain the temperature low enough.

Regarding the application of beryllium I would like to point out that the principal aluminum beam is designed to take limit load plus. However, we are stiffness critical. The weight of the beryllium caps on these frames is in the order of 6000 pounds but the total weight saving is far in excess of that. As far as structural reliability is concerned, even a full crack across the beryllium cap would have very little influence on the stiffness of the beam.

Figure F-29 shows a panel TPS which we had to add on the wing as we went from 10,200 to 10,800 ft/sec staging velocity. We have flex clips for connection to the very substantial spar cap which is really a heat sink arrangement employed to keep strength allowables high with low temperatures. It is a system of floating panels on the outside, 40" long. The material in the shock impingement area is coated columbium and the rest of the wing lower surface is HS-188.

Fortunately, with the reduction in staging velocity we can eliminate this TPS and even the body TPS, going to heat-sink tank walls. The weight penalty for going from a staging velocity of 8000 ft/sec to 11,000 ft/sec is of the order of 50,000 pounds for this booster, in addition to the dramatic effect on design criteria.

Figure F-30 shows the materials we use on this vehicle and the limitations we impose on them. With all the lessons we have paid for we are quite reluctant to jump in with new materials except maybe by first getting our feet wet with a semi-structural application. All this refers to the B-9U vehicle; the lower staging velocity vehicles are not as demanding. As a matter of fact, all these exotic materials fall more or less by the wayside with the lower staging velocities.

What I wanted to present to you was one of the most difficult vehicles that we treated and I hope that I have inspired your thinking along these lines a little bit. It is a completely new ballgame.

THRUST
STRUCTURE

ORBITER SUPPORT &
SEPARATION LINKS

CRUISE ENGINE
STOWAGE

FWD SKIRT STRUCTURE

WING/FUSELAGE
ATTACH LINKS

CREW
COMPARTMENT

M.L.G. WELL

LH_2 TANK

CANARD

TPS

INTERTANK ADAPTER

LO_2 TANK

TPS

N.L.G.
SUPPORT
STRUCTURE

FORWARD
JP TANK

ATTACH LINK

Fig. F-1

FIXED AFT
UPPER SECTION

FIXED SHELL SECTION (TYP)
(UNIDIRECTIONAL GROWTH
CAPABILITY)

FIXED LOWER
T.E. SECTION

EXPANSION SLIP JOINT
(TYP 4 PLACES)

FIXED POSITION
(TYP 3 PLACES)

LONGITUDINAL
GROWTH
DIRECTION

ASSEMBLY SPLICE

ASSEMBLY SPLICE (TYP)

NOSE SECTION

SUPPORT LINKS (TYP)

CANARD FAIRING

N.L.G. DOOR

ROLLER RECEPTICLE
FITTING (TYP)

Fig. F-2

COMPONENT	FACTOR OF SAFETY		PROOF FACTOR	APPLIED ON
	YIELD	ULTIMATE		
MAIN PROPELLANT TANKS	1.10	1.40	F.M.	MAX. OPERATING PRESSURE + DYNAMIC HEAD.
	1.10	1.40	---	LOADS (+ LIMIT PRESSURE)
	1.00	---	---	PROOF PRESSURES
PERSONNEL COMPARTMENTS	1.10	1.50	---	LOADS (+ LIMIT PRESSURE)
	1.50	2.00	1.50	MAX. OPERATING PRESSURE ONLY
	1.00	---	---	PROOF PRESSURE
WINDOWS, DOORS, HATCHES	---	3.00	2.00	MAX. OPERATING PRESSURE ONLY
AIRFRAME STRUCTURE	1.10	1.40	TBD*	BOOST & ENTRY LOADS
	1.10	1.50	TBD*	AIRCRAFT MODE LOADS
PRESSURE VESSELS	---	2.00	1.50	MAX. OPERATING PRESSURE
PRESSURIZED LINES + FITTINGS	---	2.50	1.50	MAX. OPERATING PRESSURE
ALL COMPONENTS (ABORT CONDITIONS)	1.10	1.40		ABORT LOADS (+ LIMIT PRESSURE)
ALL COMPONENTS (THERMAL STRESSES)	1.00	1.00	---	THERMAL FORCES (+ FLT. LOADS)
	1.00	1.25	---	THERMAL FORCES (ALONE)

F.M. = FRACTURE MECHANICS
ASSUMED SERVICE LIFE = 100 MISSIONS
* TO BE DETERMINED ON AN INDIVIDUAL COMPONENT BASIS DEPENDING ON DESIGN APPROACH AND COMPONENT
  CRITICALITY.

61CV6360

Fig. F-3

BOOSTER B-9U
PEAK LIMIT LOAD INTENSITIES

1   1 HR GROUND HEADWINDS
2   1 HR GROUND TAILWINDS
3   1 HR GROUND SIDEWINDS
4   LIFTOFF + GROUND HEADWINDS
5   LIFTOFF + GROUND TAILWINDS
6   LIFTOFF + GROUND SIDEWINDS
7   MAX ALPHA-Q HEADWINDS
8   MAX ALPHA-Q TAILWINDS
9   MAX BETA-Q (2400)
10   3G MAX THRUST
11   BOOSTER BURNOUT
12   BOOSTER RECOVERY
13   BOOSTER SUBSONIC GUST
14   BOOSTER 2 POINT LANDING
15   BOOSTER 3 POINT BRAKED ROLL
16   BOOSTER 2 G TAXI
17   1 DAY GROUND HEADWINDS
18   1 DAY GROUND TAILWINDS
19   1 DAY GROUND SIDEWINDS
20   TWO WEEK GROUND HEADWINDS
21   TWO WEEK GROUND TAILWINDS
22   TWO WEEK GROUND SIDEWINDS
23   BOOSTER 2.5 G POSITIVE MANEUVER
24   BOOSTER -1. G NEGATIVE MANEUVER
25   BOOSTER MAX OPERATING PRESSURE
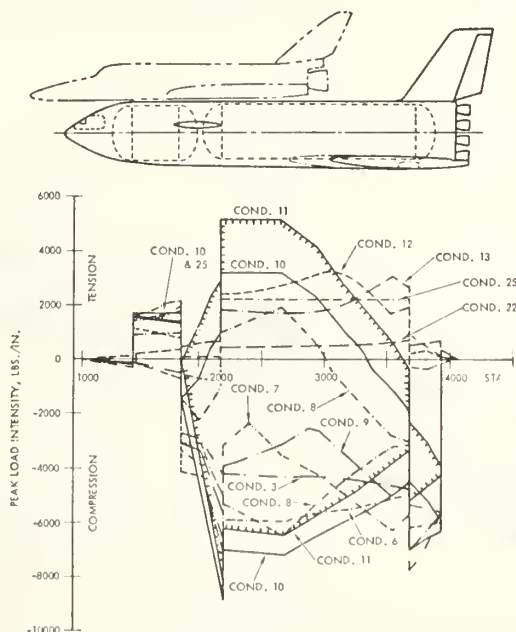
61CV6364

Fig. F-4

COMPONENT	INHERENT FEATURES	DESIGN APPROACH	PROOF TEST REQUIREMENT
LO_2 & LH_2 TANKS	NOT PRACTICAL TO DESIGN FAIL-SAFE FOR PRESSURE. STRINGERS ACT AS CRACK ARRESTORS FOR FLIGHT LOADS	SAFE-LIFE	LO_2, $\alpha = 1.23$ LH_2, $\alpha = 1.13$
WING	HIGH FAIL-SAFE CAPABILITY	FAIL-SAFE	---
CANARD BOX	SOME FAIL-SAFE CAPABILITY	SAFE-LIFE	---
CANARD PIVOT	IMPRACTICAL TO DESIGN FAIL-SAFE	SAFE-LIFE	---
TPS	IMPRACTICAL TO DESIGN FAIL-SAFE	SAFE-LIFE	---
VERTICAL TAIL	HIGH FAIL-SAFE CAPABILITY	FAIL-SAFE	---
CREW CABIN WALL	FAIL-SAFE WITH TEAR STOPPERS ONLY	FAIL-SAFE	$\alpha = 1.5$
ORBITER SUPPORT FRAMES	FAIL-SAFE WITH MULTI-ELEMENT CAPS AND WEBS	FAIL-SAFE	---

Fig. F-5

BODY - STRUCTURAL COMPONENTS

61CV6226

Fig. F-6

FWD ATTACH
PTS

AFT SUPT
TRUSS

FUSED SILICA
GLASS
WINDOWS

INTER-COMPARTMENT
BULKHEAD

CREW ACCESS HATCH

Fig. F-7

FORWARD SKIRT STRUCTURE

TPS GUIDE
ROLLER

MACHINED FRAMES -
Al ALLOY

DETAIL - TPS
GUIDE ROLLER

BOLT FLANGE
FOR ATTACHMENT
TO LO$_2$ TANK

FORWARD JP TANK
AND SUPPORTS

TPS ATTACH
LINK

N.L.G. SUPPORT
STRUCTURE - TITANIUM
TUBES

BUILT-UP SHELL
STRUCTURE
Al ALLOY

Fig. F-8

INTEGRAL
STIFFENER

RECIRCULATION FLOW
DIVIDER

BAFFLE

RADIAL
SUPPORT BEAMS

LO₂ DUCT
FLANGE (4)

ORBITER FORWARD
ATTACH BULKHEAD

ANTI-VORTEX
BAFFLE
INSTALLATION

DOME
STIFFENERS

BOLT FLANGE
DETAIL-TYPICAL
BOTH ENDS

SUMP

SUMP AND ANTI-VORTEX
BAFFLE DETAIL

ACCESS
PANEL

LAUNCH TOWER
ATTACH PROVISION

ANTI-SLOSH BAFFLES (8)

MILLED INTEGRALLY
STIFFENED SKINS-
WELDED ASSEMBLY

MATERIAL: 2219 Al
ALLOY

STABILIZING
FRAMES

TYPICAL STABILIZING
FRAME DETAIL

Fig. F-9

INTERTANK ADAPTER STRUCTURE

ORBITER DRAG LINK
ATTACH FITTINGS (2)
TITANIUM

ACCESS DOOR

CANARD SUPPORT
BEARING - OUTBOARD
(BOTH SIDES)

ORBITER SUPPORT FRAME

CANARD SUPPORT
FRAMES (2)

CANARD SUPPORT
BEARING - INBOARD
(BOTH SIDES)

INTERMEDIATE
FRAMES (2)

LO₂ SUPPLY LINE
EXIT (BOTH SIDES)

CANARD
ACTUATOR ACCESS
(BOTH SIDES)

LO₂ TANK ATTACH
FLANGE

ORBITER SUPPORT FRAME

STIFFENED INTERTANK
SKIN

LO₂ SUPPLY LINE
EXIT (BOTH SIDES)

Fig. F-10

STABILIZING FRAMES
(TYPICAL)

MILLED INTEGRALLY
STIFFENED SKINS -
WELDED ASSEMBLY

ORBITER LINK ATTACH FITTINGS

ORBITER AFT ATTACH BULKHEADS

FRAME STABILIZING
LINKS (TYPICAL)

BOLT FLANGE
BOTH SIDES

ACCESS
PANEL

WING ATTACH
BULKHEAD (LINK
FITTING)

MLG WELL ATTACH
BULKHEADS

TPS ATTACH
PROVISION

TPS SUPPORT
FRAMES

INBOARD RIB - WING

MATERIAL: 2219 Al ALLOY

WING DRAG LINK

VERTICAL
LINK

DETAIL OF WING - FUSELAGE ATTACHMENT

Fig. F-11

MAIN LANDING GEAR SUPPORT STRUCTURE

WHEEL WELL TO
LH$_2$ TANK
DRAG LINK

BULKHEAD
STATION 3295

SIDE LOAD LINK

CARRY-THROUGH
OR "SADDLE" BEAMS
(2) TITANIUM

FORWARD LINKS
VERTICAL & SIDE LOADS

VERTICAL LOAD
LINK

DRAG LONGERON
ON LH$_2$ TANK

M.L.G. MAIN
TRUNNION FITTING
(2 PER SIDE)

BULKHEAD
STATION 3020

M.L.G. DRAG STRUT TO
WHEEL WELL FITTING

STRUCTURAL
WHEEL WELL
ENCLOSURE
Al ALLOY
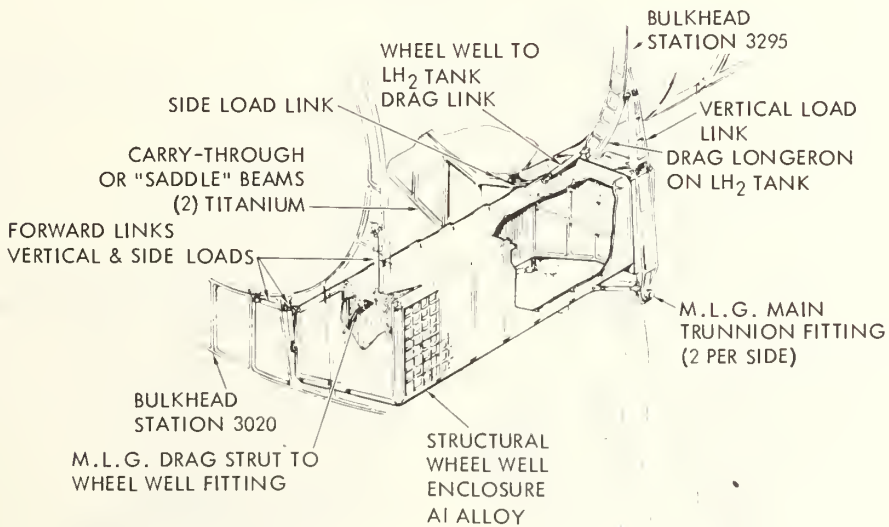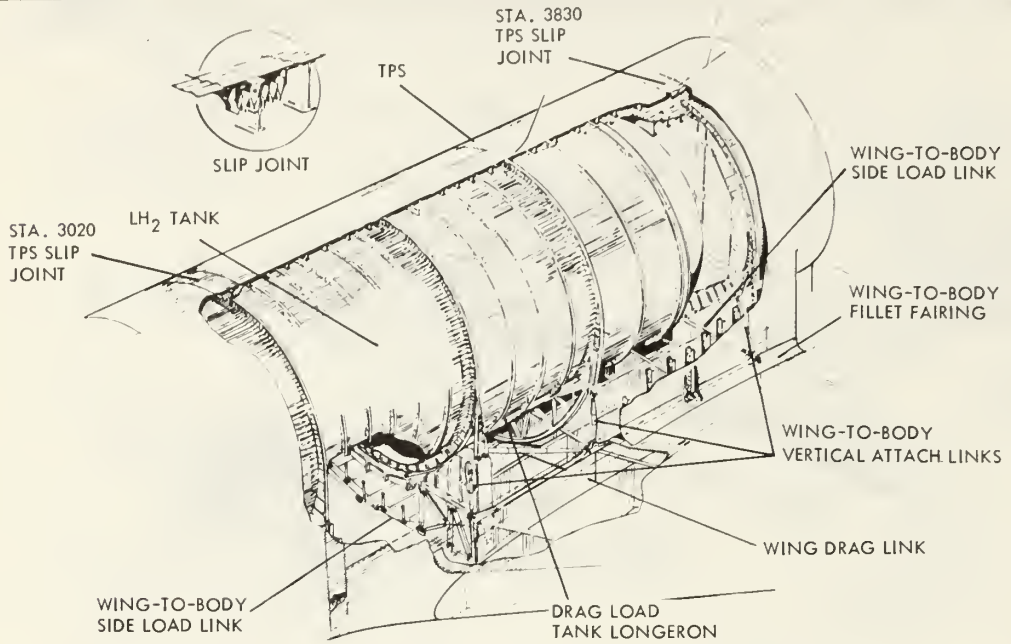
Fig. F-12

Fig. F-13

CRYOGENIC INSULATION LH₂ TANK
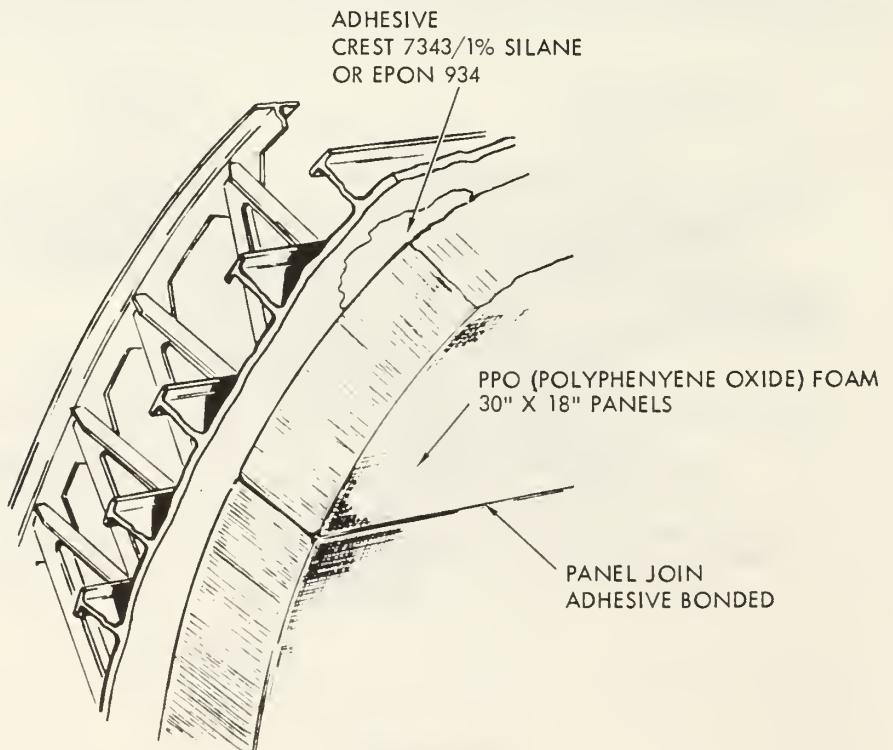
Fig. F-14

THRUST STRUCTURE

BORON/ALUMINUM COMPOSITE
THRUST POSTS (12)

TITANIUM SKIN

VERTICAL STABILIZER
ATTACH FITTINGS - TITANIUM

BORON/ALUMINUM
COMPOSITE STRINGERS

HOLDDOWN
FITTING (4)
TITANIUM

THRUST SECTION -
LH$_2$ TANK ATTACH
FLANGE

TITANIUM - BORON/Al
COMPOSITE REINFORCED
THRUST BEAMS (4)

TITANIUM
BULKHEADS

BORON/ALUMINUM
COMPOSITE TRUSSES (TYP)

WING ATTACH
FITTING - TITANIUM

Fig. F-15

BASE HEAT SHIELD

RENE' 41 SUPPORT CHANNELS

TITANIUM CORRUGATED SKIN

RENE' 41 BASE BULKHEAD

COATED COLUMBIUM CORRUGATED PANEL
(CENTER FOUR PANELS ONLY)

RENE' 41 CORRUGATED PANELS

RENE' 41 REINFORCING RING

BERYLIUM TRUSS TUBES (TYP)

Fig. F-16

LEADING EDGE COATED COLUMBIUM

SKINS TITANIUM CORRUGATED

ACPS THRUSTERS

SIDE LOAD LINKS

BODY SUPPORT ATTACH FITTINGS TITANIUM

SPAR TRUSS TITANIUM

SPAR TITANIUM CORRUGATED

ELEVON

MICHOUD MFG. BREAK

ABES BAYS

FORMERS TITANIUM

RIB TITANIUM

WING TIP DETACHABLE

JP FUEL TANK

Fig. F-17

WING STRUCTURE
MULTISPAR CORRUGATED SKIN CONCEPT

MULTISPAR PRIMARY BENDING STRUCTURE

TORSION BOX

TITANIUM WEB

TITANIUM SPAR CAP

300°F MAX.

1,350 °F   SKIN (INCONEL 718)

ACCOMMODATES SKIN EXPANSION RELATIVE TO SPAR

Fig. F-18

VERTICAL STABILIZER

SECTION A-A

TIP

INTEGRALLY STIFFENED SKIN (Ti)

FRONT SPAR

CENTER SPAR

REAR SPAR

INTERNAL STRUCTURE TITANIUM

TRAILING EDGE

RIB

RUDDER

LEADING EDGE INCONEL 718

CORRUGATED SPAR WEB

FORMER

SPAR-TO-THRUST STRUCTURE ATTACH FITTINGS
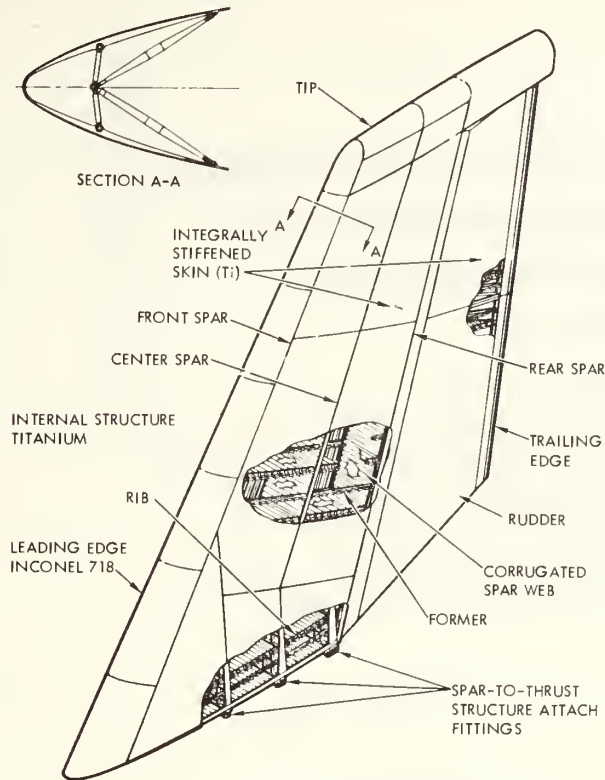
Fig. F-19

CANARD STRUCTURE

ACTUATOR TRUSS SUPPORT SYSTEM

VENT HOLES

PIVOT TUBE

ACTUATORS

CARBON-CARBON

LEADING EDGE TRUSS RIBS
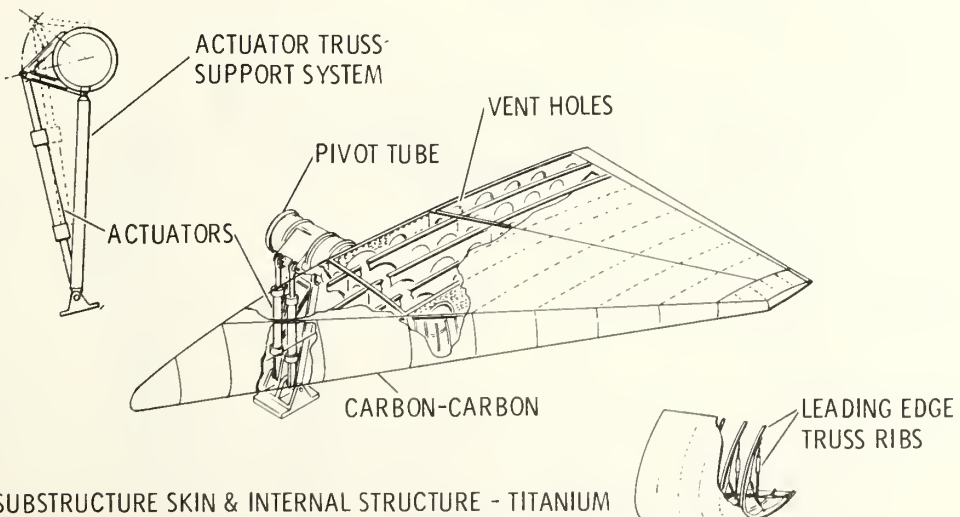
- SUBSTRUCTURE SKIN & INTERNAL STRUCTURE - TITANIUM
- UPPER SURFACE TPS OUTER SKIN — COATED COLUMBIUM
- LOWER SURFACE TPS OUTER SKIN — RENE' 41

Fig. F-20

- LIFE 100 FLIGHTS, NO REFURBISHMENT
- AIRLOAD/TEMPERATURE HISTORY
- ACOUSTICAL ENVIRONMENT
- VIBRATION
- PANEL FLUTTER
- FACTOR OF SAFETY = 1.4
- FATIGUE FACTOR OF 4 ON CYCLES
- CREEP FACTOR OF 4 ON LIFE
- TOTAL CUMULATIVE CREEP - 0.2%
- LEAKAGE 350 IN.^{2} EQUIVALENT

Fig. F-21

PANEL PRESSURES, TPS SHELL

STATION   1000   1458   2040   2510   3800

$\frac{X}{L}$ → 0   0.163   0.342   0.506   0.938

LIMIT PRESSURES (PSI)

LOAD CONDITION	STATION 1485	STATION 2040	STATION 2510	STATION 3800
MAXIMUM αq ASCENT $p_\infty = 6.5$ psi $P_{internal} = 6.0$ psi	-1.30 -1.30 -1.20	-1.00 -0.90 -.70	-0.90 -0.40 -0.40	-1.50 -0.70 -0.90
MAXIMUM g ENTRY $p_\infty = 0$ psi $P_{internal} = 0$ psi	0 0 +2.20	+0.03 +0.10 +1.95	0 0 +2.00	0 0 +1.80
MAXIMUM αq ENTRY $p_\infty = 0$ psi $P_{internal} = 0$ psi	+0.08 +0.30 +2.10	+0.05 +0.12 +1.50	+0.03 +0.10 +1.65	+0.02 +0.10 +1.10

(+) DENOTES CRUSHING PRESSURE
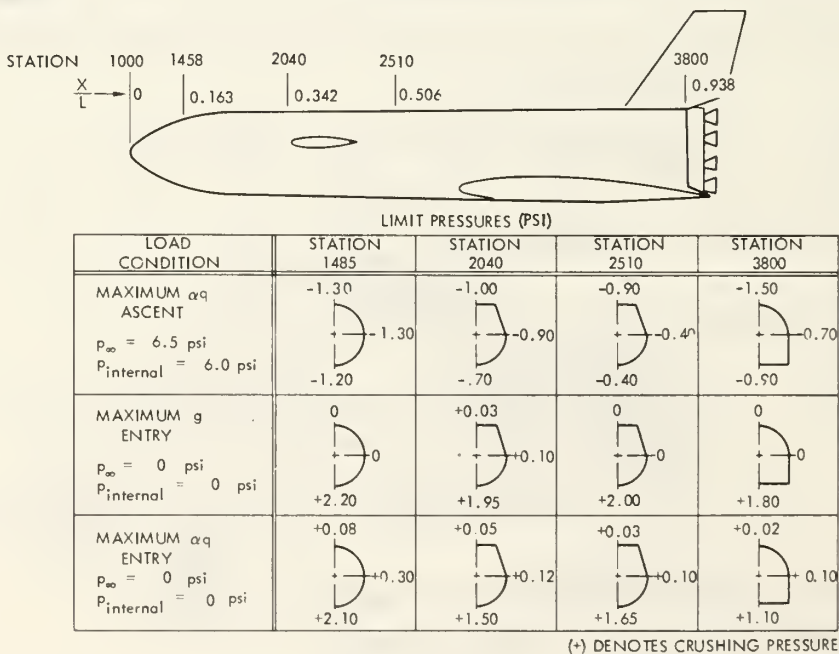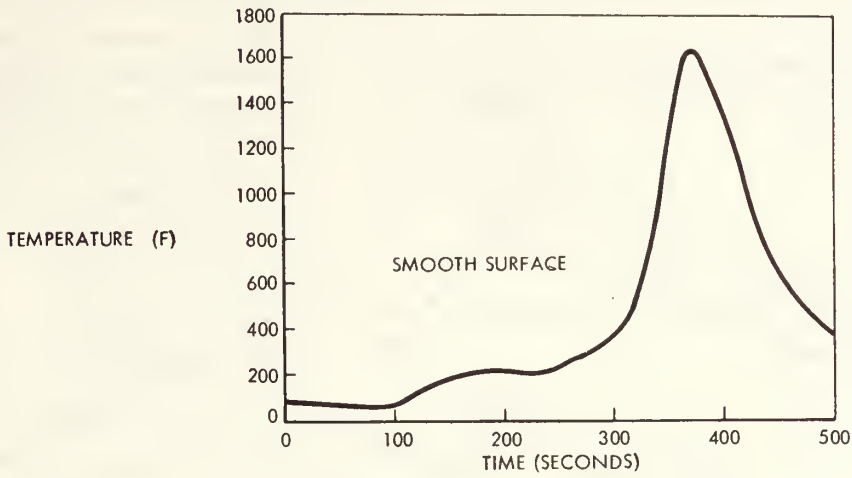
Fig. F-22

BODY LOWER SURFACE, Θ= 0 DEGREES

Fig. F-23

TPS TEMPERATURE VS. THICKNESS
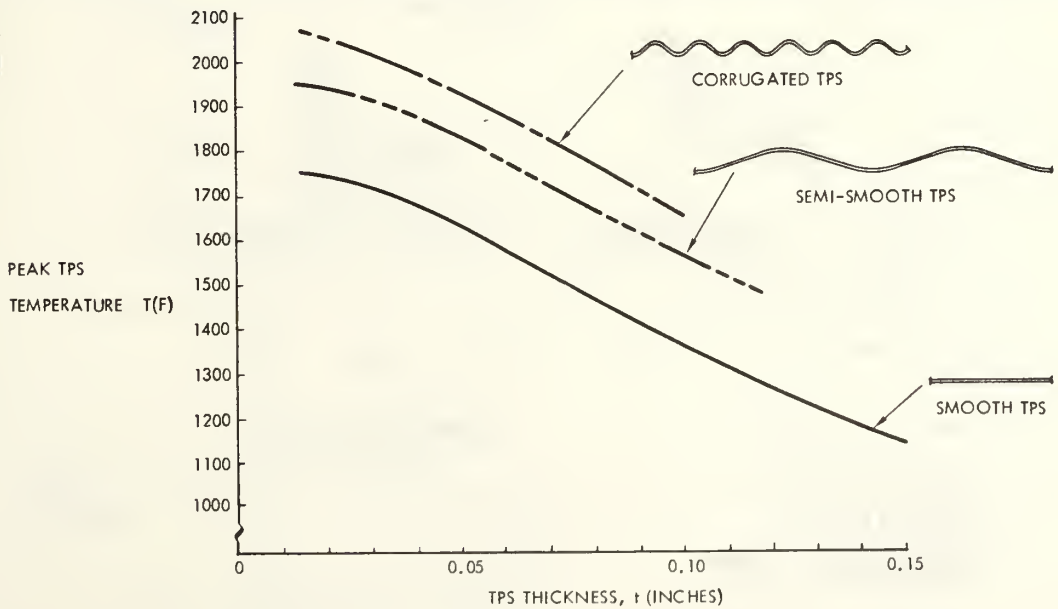
Fig. F-24
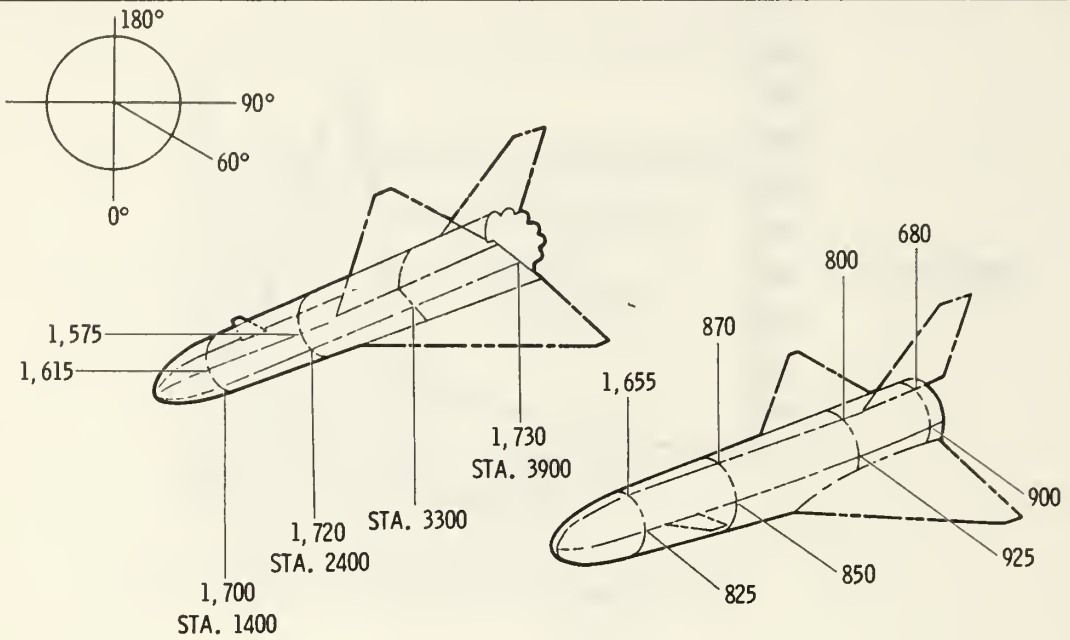
Fig. F-25
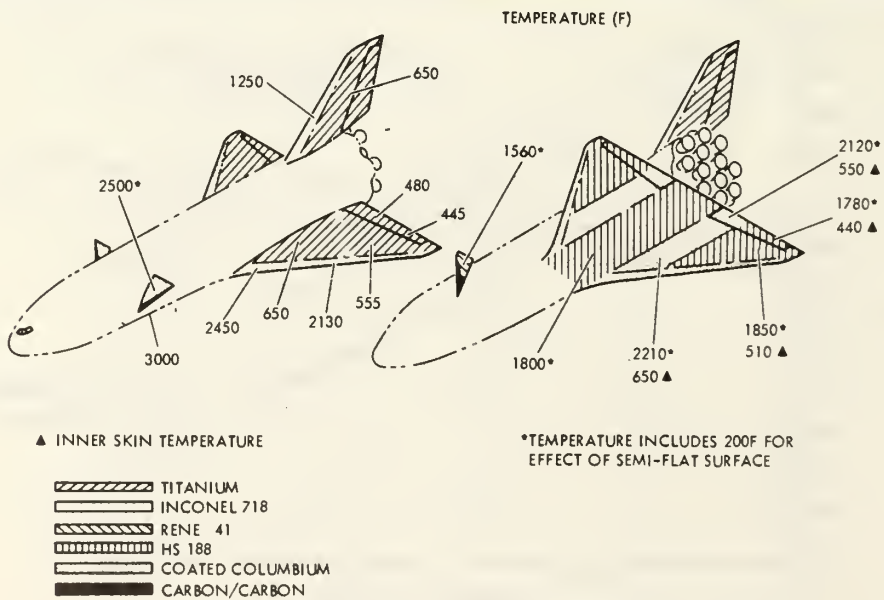
Fig. F-26

LOCATION ON TPS	PANEL CONFIGURATION	MATERIAL	SHEET THKNESS (F)	TEMP ◁ (F°)	MAX. THERMAL GRADIENT STRESS (PSI)	TOTAL STRESS (PSI)	ALLOW STRESS (PSI)
LOWER SURFACE	1.25 2.50	RENE' 41	.016	1,750°	39,200	43,700	4,100
	1.25 2.50	RENE' 41	.055	1,600°	35,700	37,700	13,700
	.40 6.00	RENE' 41	.055	1,600°	0	11,800	13,700
SIDE	1.00 4.60	INCONEL 718	.035	1,180°	44,500	59,000	64,500
SIDE & TOP	1.00 4.60	TITANIUM 6AL-4V	.030	695°	6,000	17,500	18,500

◁ INCLUDES EFFECTS OF HEAT SINKING AND SURFACE GEOMETRY

Fig. F-27

BERYLLIUM CAP
ALUMINUM FRAME
MECHANICAL SKIN FASTENERS
INTEGRALLY MACHINED STIFFENER
BERYLLIUM CAP
SEMI-SMOOTH RENE' SKIN

SECTION A-A

9 IN.
STANDARD FRAME

SUB-FRAME

40" 20"

Fig. F-28

SPAR CAP

CORRUGATED
INNER SKIN

INSULATION

OUTER SKIN
(SEMI-FLAT)

STAND-OFF

UP

FWD   INBD

40" PANEL
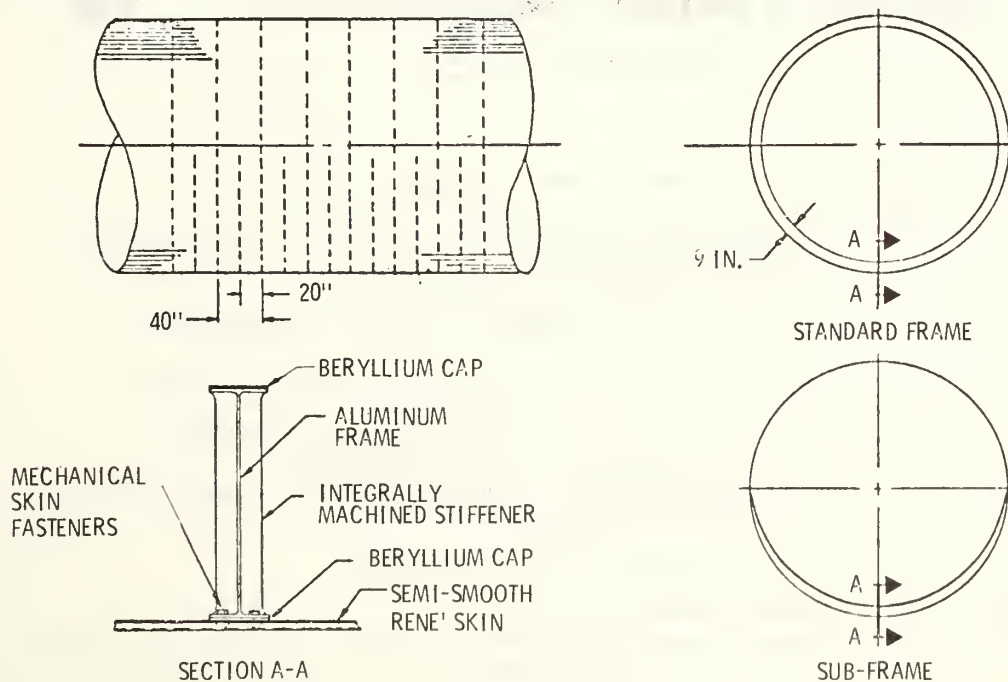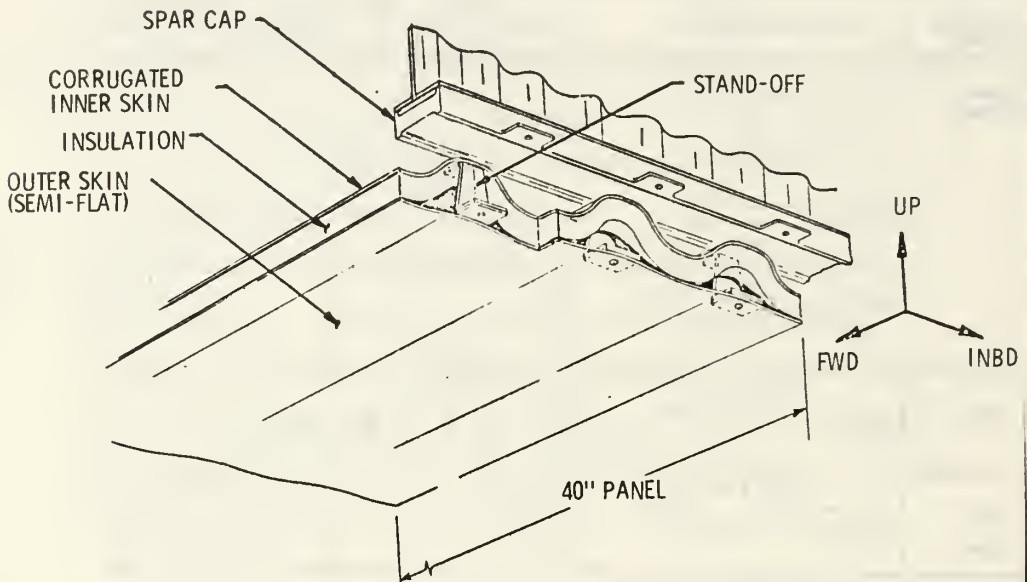
Fig. F-29

BOOSTER TPS MATERIAL TEMPERATURE LIMITS

MATERIAL	TEMP. (°F)	RATIONALE
TITANIUM	800/ 650	STRESS CORROSION CRACKING LIMITS USE AT TEMP. ABOVE 650°F TO VERY SHORT TIME & LOW STRESS. HIGHER LIMIT OF 800°F MAY BE CONSIDERED FOR BOOSTER COMPONENTS WHERE TIME & STRESS CONDITIONS PERMIT
INCONEL 718	1,350	1,350°F IS AGE TEMP. OPERATION ABOVE THIS TEMP. SEVERELY DEGRADES MATERIAL
RENE' 41	1,650	ABOVE 1,600°F, INTEGRANULAR OXIDATION OCCURS. TIME ABOVE THIS ON BOOSTER IS SHORT ENOUGH TO BE ACCEPTABLE
HAYNES 188	1,900	CREEP LIMITED. TIME AT TEMP. AFFECTS TOTAL CREEP. BOOSTER ACCEPTABLE BECAUSE TIME IS APPROXIMATELY 1 MINUTE PER MISSION
TD NiCr	2,200	LIMITED BY CREEP & HIGH TEMP. STRAIN CAPABILITY. TIME AT TEMP. & LOADS OF BOOSTER PERMIT HIGHER TEMP. APPLICATION
COLUMBIUM 752	2,500	2,500°F IS UPPER LIMIT FOR COATING REUSABILITY
CARBON/ CARBON	3,000	BASED ON PLASMA-ARC TESTS

Fig. F-30

STRUCTURAL DESIGN CONSIDERATIONS FOR
ADVANCED AIRCRAFT

J. E. Fischler
Douglas-Long Beach

Only the figures and an abstract are available.

Abstract: Structural Engineers must continue to investigate ways
to save structural weight without losing structural
integrity. Two computing programs were used to
determine a basis for comparing advanced materials and
types of construction with existing aluminum structure.
One computing program determines the rate of crack
growth from the material characteristics (K_C = critical
stress intensity factor, R = stress range, $\triangle K$ = stress
intensity factor, B and n = material characteristics
and type of construction, and R_{ct} = stress ratio of an
unstiffened panel to one with stiffeners and/or crack
stoppers). A second program solved for the probability
of structural failure using the mean 1 g stress and
coefficient of variation of mean load (dependent on the
type of mission), mean strength and coefficient of
variation of mean strength, and residual strength which
decreased as a function of the crack length and panel
stiffening at any time after first flight. Ways were
shown to calculate the probability of initiation of a
crack and its percentage of total fatigue damage. For
ease of comparison, and reduction of computer time,
all the materials compared were based on an undetected
crack of 0.2 inches initially. The theoretical results
were compared to actual tests on a 96 inch wide stiffened
panel. Modifications to the theoretical coefficients
were recommended due to practical considerations of the
slowdown in crack growth when the crack tip reaches an
attachment, when a stiffener fails, when the fracture
toughness, K_c and $\triangle K$ values vary with crack length, when
the value of R_{ct} varies with stiffener area and crack
length, and when the stress spectra has occasional
high stress cycles that modify the plastic zone ahead
of the crack tip and actually <u>slows down</u> the rate of
crack growth for the average cyclic stresses.

The materials compared were aluminum (2024-T3, 7075-T6), titanium and graphite epoxy. The great weight advantage of graphite-epoxy at as low as 10,000 psi 1 g stress and the low probability of failure allowed many more flights and longer inspection intervals (thus reducing Maintenance Costs).

Desirable future analysis methods, material characteristic and type of construction research, and other structural research plans were illustrated as areas for Structural Development to reduce the structural weight and improve the Structural Reliability (Lower the Probability of Structural Failures).

One of the most desirable results of the study are the charts that have been prepared that represent over two hundred data points. These charts give the Structural Analyst the opportunity to compare materials at different stress levels, mission load variations, mean strength variations, number of flights, structural efficiency, residual strength, and variation of material parameters all versus the Probability of Failure. These overall views of various materials and types of construction can be very useful for choosing a material and also determining what future research is essential for saving structural weight and improving structural reliability.

DIRECT OPERATING COST = CENTS/SEAT-MILE
RETURN ON INVESTMENT = PROFIT PER YEAR/COST

$$D.O.C. = \frac{\text{FLIGHT OPER COSTS + DEPRECIATION + FLIGHT INSURANCE + MAINTENANCE + FUEL COSTS}}{\text{RANGE} \times \text{PAYLOAD}}$$

$$\text{RETURN ON INVESTMENT} = \frac{\text{PROFIT PER TRIP} \times \text{NUMBER TRIPS PER YEAR}}{\text{ORIGINAL COST PER AIRCRAFT}}$$

$$\text{RANGE} = 2.3 \frac{V}{c'} (L/D)_{MAX} \text{ LOG}_{10} \frac{W_0}{W_1}$$

W_0 = WEIGHT AT INITIAL CRUISE

W_1 = WEIGHT AT FINAL CRUISE

Fig. G-1
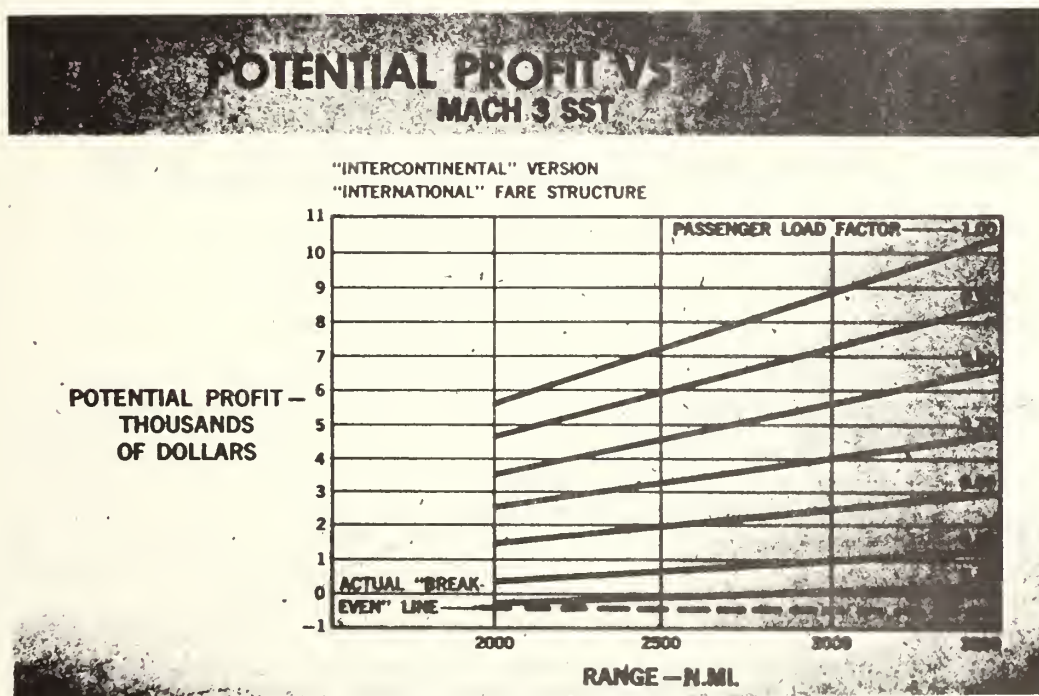
Fig. G-2

DISTRIBUTIONS OF COST AND WEIGHT
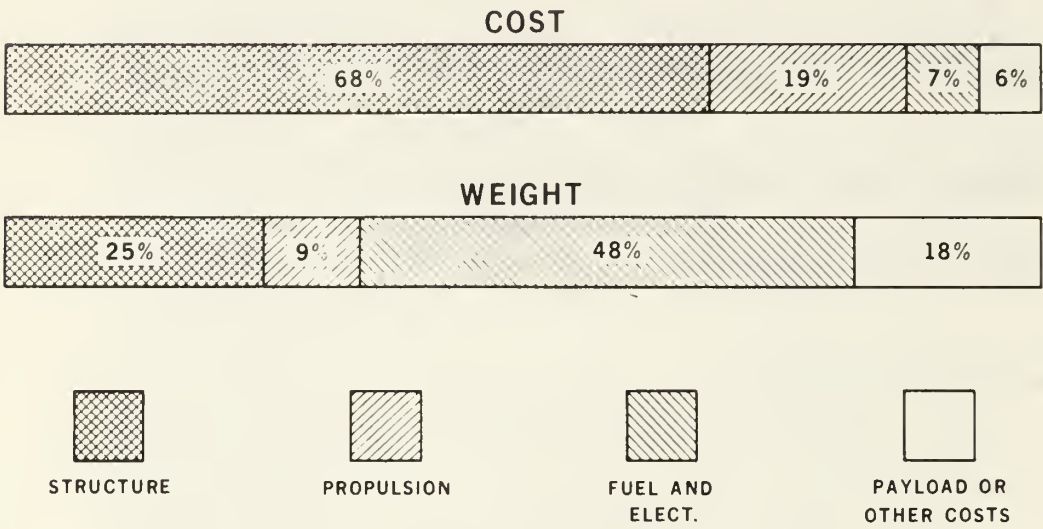
Fig. G-3

RANGE CHART
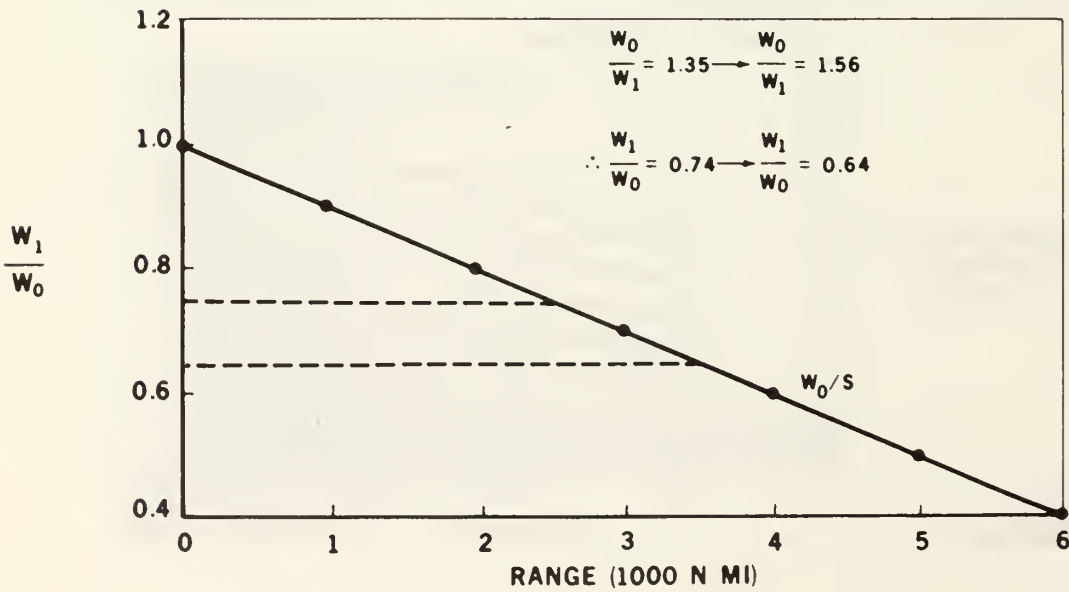
Fig. G-4

PAYLOAD vs GROSS WEIGHT
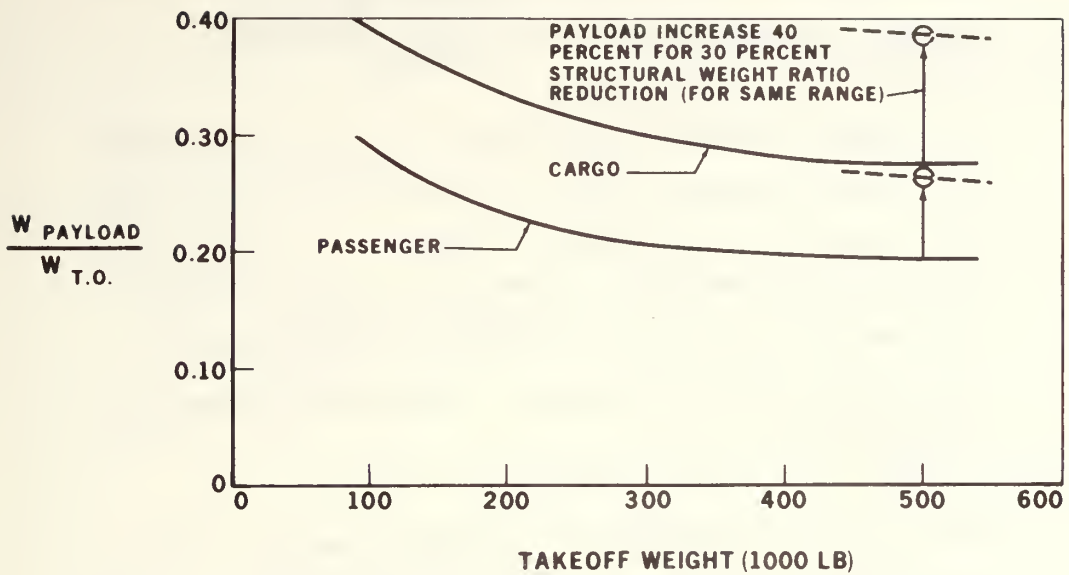
Fig. G-5

NEW VEHICLE COST
WITH CHANGE IN STRUCTURAL COST

ASSUME 30 PERCENT STR WGT DECREASE = 60 PERCENT STRUCTURAL COST INCREASE (COMPOSITE STRUCTURE)

30 PERCENT STR WGT → 40 PERCENT PAYLOAD GAIN

$$\text{NEW COST} = \left[\frac{1.6 \times 0.68}{1.30} - 0.68 \right] = 15.6 \text{ PERCENT COST INCREASE}$$

ASSUMED VEHICLE COST = 25 MILLION

1.156 × 25M = 28.9 MILLION = (NEW COST)

Fig. G-6

RETURN ON INVESTMENT

$2000/FLIGHT AT 60 PERCENT LOAD FACTOR (SST)

$$\text{RETURN ON INVESTMENT} \atop (45\% \text{ BREAKEVEN L.F.}) = \frac{\$2000 \times 4 \text{ FLTS/DAY} \times 300 \text{ FLTS/YR}}{25,000,000}$$

R.O.I. = 9.6 PERCENT PER YEAR

NEW AIRCRAFT IS 30 PERCENT LESS STR. WGT.

$$\text{NEW STR. WGT. FRACTION} = \frac{25\% \text{ W}}{1.3} = 19.2\% \text{ W}$$

LET ALL STR. WGT. DIFF. BE ADDED TO PAYLOAD

OLD PAYLOAD FOR BREAKEVEN = 0.45 x 18% W = 8.1% W

$$\text{RATIO INCREASE IN REVENUE} = \frac{8.1 + (25\% \text{ W} - 19.2\% \text{ W})}{8.1} = 1.72$$

STRUCTURAL COST HAS GONE UP 60% = 15.6% FOR TOTAL AIRCRAFT

NEW BREAKEVEN = X

$$(0.60 \text{ L.F.} - X) = (0.60 - 0.45) \frac{1.72}{1.156} \therefore X = 0.377$$

NEW PROFIT/FLT. AT 60 PERCENT L.F. = $2980/FLT (USING 60% − 37.7%)

$$\text{NEW RETURN ON INVESTMENT} \atop (\text{AT 3.7\% BREAKEVEN L.F.}) = \frac{2980 \times 4 \text{ FLT/DAY} \times 300}{28,900,000}$$

NEW R.O.I. = 12.4 PERCENT/YEAR

PERCENT IMPROVEMENT = 12.4/9.6 − 1 = 30 PERCENT

Fig. G-7

MAINTENANCE GAINS

(1)   MAINTENANCE = TWICE ORIGINAL COST OR 25 PERCENT OF D.O.C.

(2)   ASSUME 20 PERCENT OF MAINTENANCE COST FOR ACCESS, INSPECTION, AND REPAIRING STRUCTURE AT SCHEDULED INSPECTIONS.

(3) ASSUME GOAL IS TO DOUBLE INSPECTION INTERVAL (i.e., 9,200 FLT TO 18,400 FLT)

(4) PROGRAM COST REDUCTION = $\frac{20\%}{2}$ x 2 x 25M = 5 MILLION

D.O.C. COST REDUCTION $\cong \frac{20\%}{2}$ x 25% = 2.5%

Fig. G-8

I-112

GAIN IN DIRECT OPERATING COST
STRUCTURE WEIGHT SAVING AND DOUBLE INSPECTION INTERVAL

$$\Delta \text{D.O.C.} \approx 1 - \frac{\text{OLD CONFIGURATION}}{\text{NEW CONFIGURATION}}$$

	OLD CONFIGURATION $\left(\begin{array}{l}\text{PRICE} = 25M \\ \text{MAINT} = 50M\end{array}\right)$	NEW CONFIGURATION $\left(\begin{array}{l}\text{PRICE} = 28.9M \\ \text{MAINT} = 45M\end{array}\right)$
DEPRECIATION/TRIP (12 YR) AND 14,400 TRIPS	$1730	$2010
MAINTENANCE/TRIP	$3480	$3120
FUEL COST/TRIP AT $0.01493/N MI	$4300	$4300
FUEL = 0.48 WGT		
WGT = 600,000 LB		
RANGE x PAYLOAD	1.0	1.4 (40 PERCENT BETTER FOR 30 PERCENT STR WGT SAVING)
	$9510	$9430

$$\Delta \text{D.O.C.} = 1 - \frac{\dfrac{9510}{9430}}{1.4} = 1 - 1.41$$

$$\Delta \text{D.O.C.} = -0.41$$

Fig. G-9

CRACK LENGTH AS A PERCENTAGE OF FATIGUE LIFE
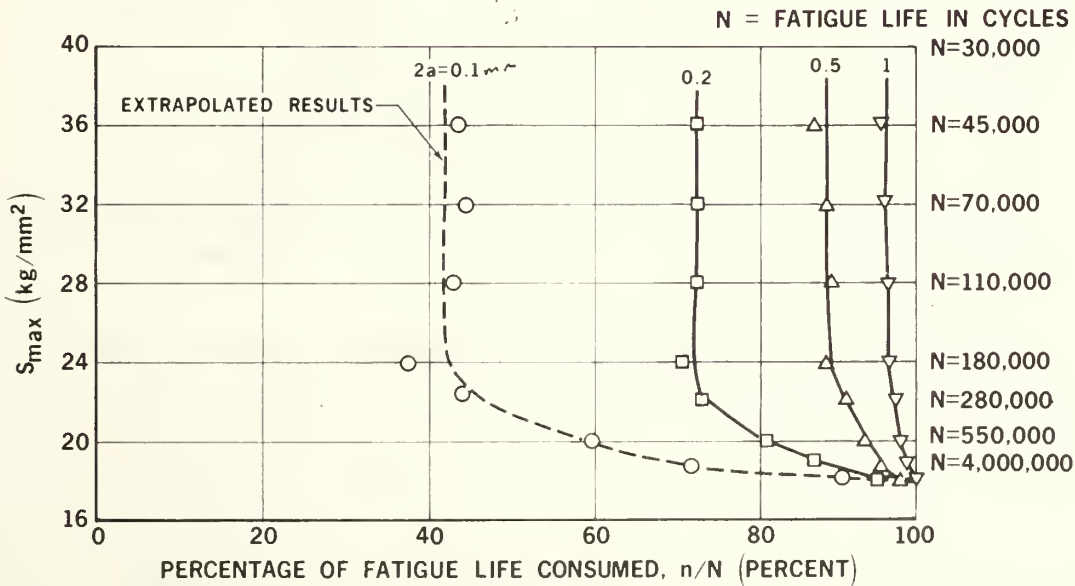
Fig. G-10

A SAFETY GOAL (P_s, T_l 10^{10})   C SUPERSONIC SPECTRUM   E SUPERSONIC SPECTRUM
B SUBSONIC SPECTRUM t_l 1500 hrs, R   4000 hrs t_l = 3000 hrs, R = 4000 hrs
   t_l 3000 hrs, R   10 000 hrs   D SUBSONIC SPECTRUM   F T_c = 50,000 hrs
                                          t 3000 hrs, R   4000 hrs   G T_c = 30,000 hrs
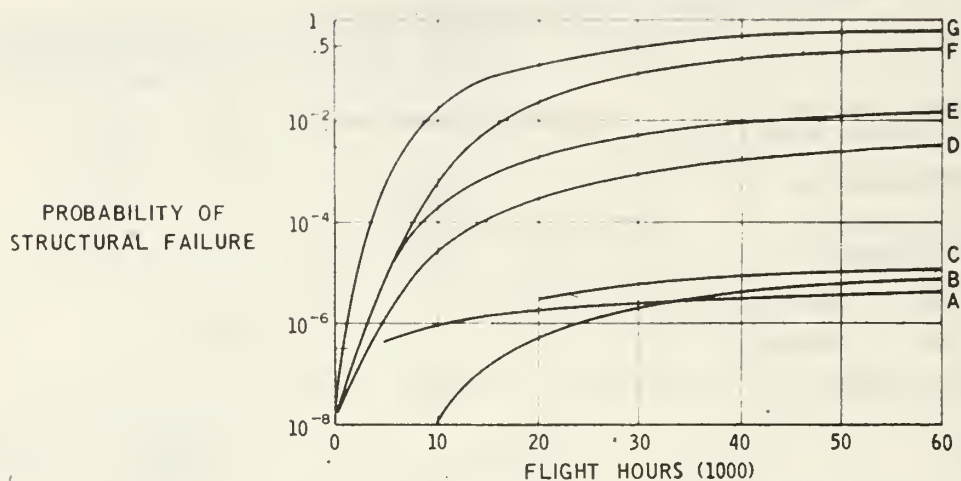
FIGURE 11. PROBABILITY OF STRUCTURAL FAILURE OF FATIGUE
CRITICAL WING COMPONENT VERSUS FLIGHT HOURS

Fig. G-11

RELIABILITY
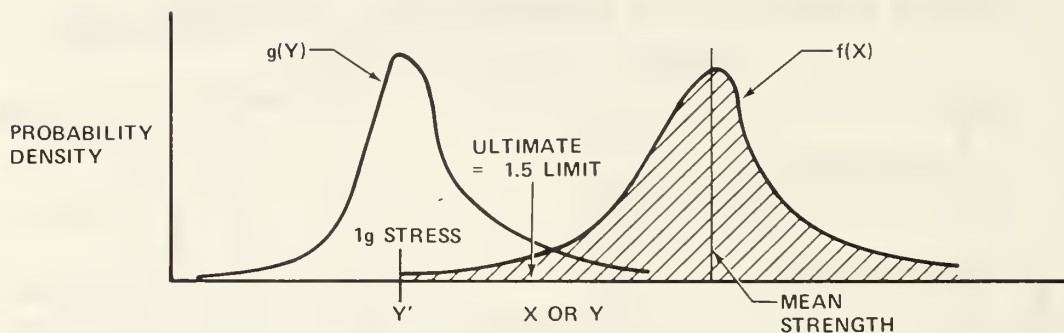
R   =   RELIABILITY

R   =   pR $(X > Y)$ = PROBABILITY THAT THE STRENGTH X IS GREATER THAN
THE APPLIED STRESS Y

$$R = \int_{-\infty}^{\infty} G(X)\, f(X)\, dX$$

$$G(X) = \int_{-\infty}^{X} g(Y)\, dY$$

IF Y' IS <u>NOT</u> A RANDOM VARIABLE BUT <u>ALWAYS</u> OCCURS (1g STRESS), THEN
R   =   pR $(X > Y)$ = SHADED AREA UNDER f(X)

PR71-GEN-2C

Fig. G-12

I-114

PROBABILITY
DENSITY
FUNCTION
OF LOAD
OR STRENGTH

LOAD OR STRENGTH
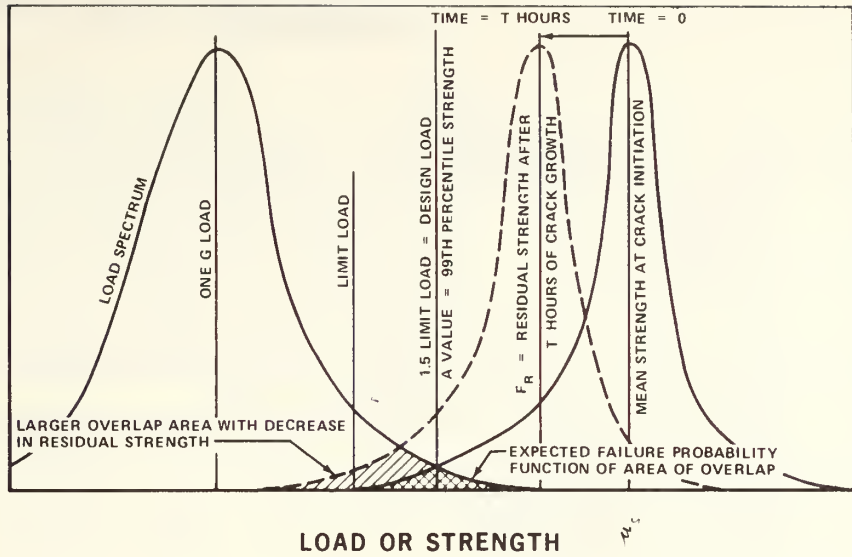
PR71-GEN-20448

Fig. G-13

PROBABILITY OF STRUCTURAL FAILURE
DEPENDS ON:

$$\frac{\mu_S}{\mu_L} = \frac{\text{MEAN STRENGTH IN PSI}}{\text{MEAN LOADING IN PSI (ONE g STRESS)}}$$

$$\frac{\sigma_L}{\mu_L} = \frac{\text{STANDARD DEVIATION OF LOADING}}{\text{MEAN LOADING (ONE g)}} \text{ (MEASURE OF SPECTRA RANGE)}$$

$$\frac{\sigma_S}{\mu_S} = \frac{\text{STANDARD DEVIATION OF STRENGTH}}{\text{MEAN STRENGTH}}$$

$$F_R = \text{RESIDUAL STRENGTH}$$

$$F_R = \frac{K_C R_{CT}}{C \sqrt{W \, \text{TAN} \frac{\pi a}{W}}}$$

K_C = FRACTURE TOUGHNESS = VARIES WITH MATERIAL, DIRECTION OF LOADING, CRACK STOPPERS = (TYPE OF CONSTRUCTION); SIZE OF CRACK TO PANEL WIDTH, THICKNESS.

R_{CT} = RATIO OF STRESS IN THE REGION OF THE CRACK TIP IN AN UNSTIFFENED PANEL TO THAT IN A STIFFENED PANEL

C = FINITE PANEL WIDTH CORRECTION FACTOR

W = PANEL WIDTH

a = HALF CRACK LENGTH

Fig. G-14

PROBABILITY OF STRUCTURAL FAILURE
ALSO DEPENDS ON:

$\dfrac{da}{dN}$ = RATE OF CRACK GROWTH PER CYCLE

$\dfrac{da}{dN}$ = $\dfrac{B \Delta K^n}{(1-R) K_c - \Delta K}$

ΔK = $\dfrac{\sigma_{MAX}}{R_{CT}} (1-R) C \sqrt{W \, TAN \, \dfrac{\pi a}{W}}$ = RANGE OF STRESS INTENSITY FACTOR

B = VARIES WITH MATERIAL, TYPE OF CONSTRUCTION, LOCATION OF CRACK TO ATTACHMENTS, TYPE OF ATTACHMENTS

R = $\dfrac{\sigma_{MIN}}{\sigma_{MAX}}$ = $\dfrac{MINIMUM \; STRESS}{MAXIMUM \; STRESS}$

K_c = FRACTURE TOUGHNESS

Fig. G-15

PROBABILITY OF FAILURE AFTER A CRACK OCCURS

$$P \, (FAILURE) = \frac{1}{2} \, erfc \left[\frac{\dfrac{\mu_S}{\mu_L} - 1}{\sqrt{2} \left[\left(\dfrac{\sigma_L}{\mu_L} \right)^2 + \left(\dfrac{\sigma_S}{\mu_S} \right)^2 \left(\dfrac{\mu_S}{\mu_L} \right)^2 \right]^{1/2}} \right]$$

WHERE: $\dfrac{\mu_S}{\mu_L}$ = $\dfrac{1.5 \, L_S}{\mu_L \left(1 - 2.575 \dfrac{\sigma_S}{\mu_S} \right)}$

L_S = LIMIT STRENGTH

FACTOR OF SAFETY = 1.5 BETWEEN THE DESIGN LIMIT LOAD AND THE 99TH PERCENTILE VALUE OF ULTIMATE STRENGTH

Fig. G-16

PROBABILITY OF STRUCTURAL FAILURE vs
STRENGTH, LOAD AND NUMBER OF FLIGHTS
ALUMINUM WING
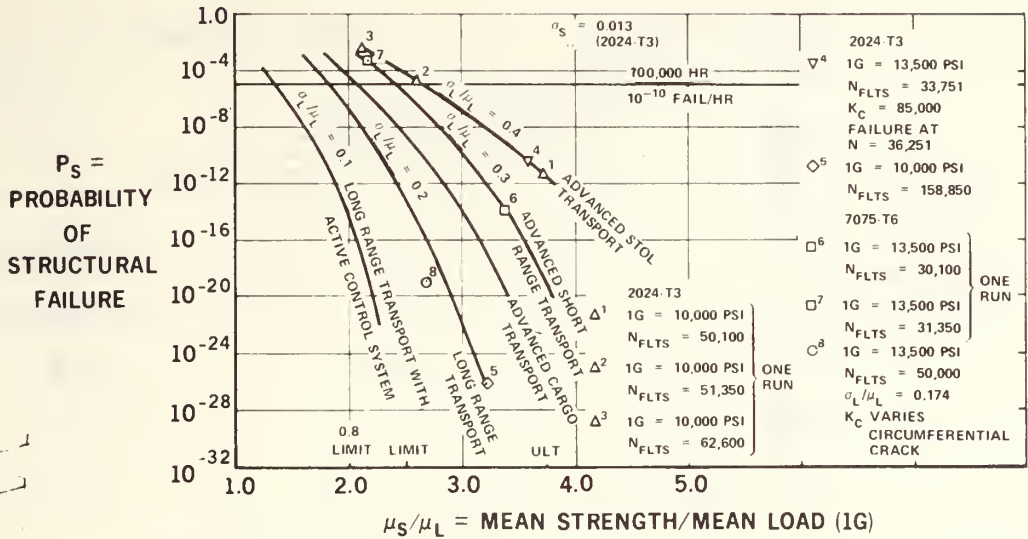
Fig. G-17

PROBABILITY OF STRUCTURAL FAILURE vs
STRENGTH, LOAD AND NUMBER OF FLIGHTS
TITANIUM WING
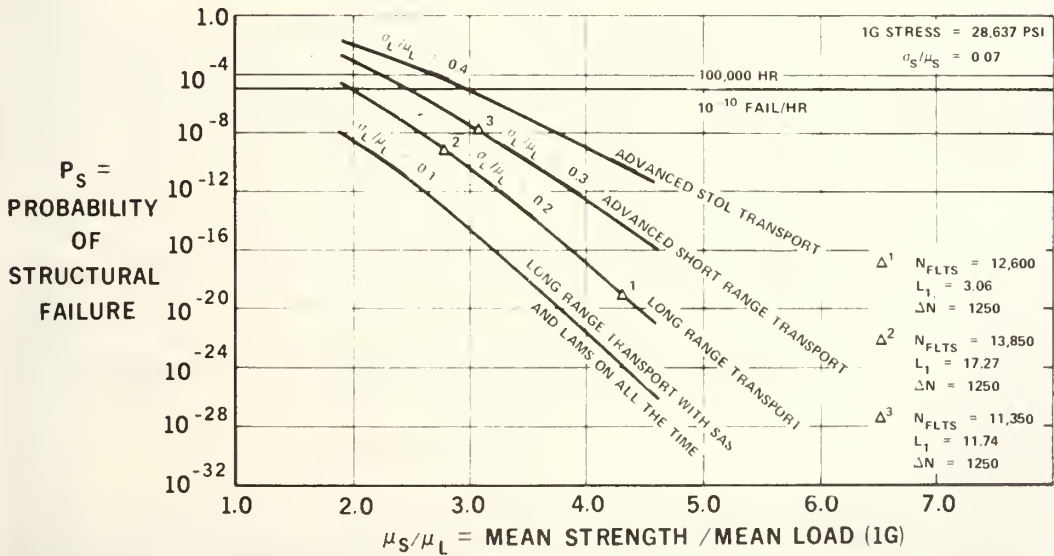
Fig. G-18

PROBABILITY OF STRUCTURAL FAILURE vs STRENGTH, LOAD AND NUMBER OF FLIGHTS

GRAPHITE/EPOXY

Fig. G-19

FIGURE 20 MEAN STRESS VERSUS FATIGUE LIFE FOR SUPERSONIC TRANSPORT
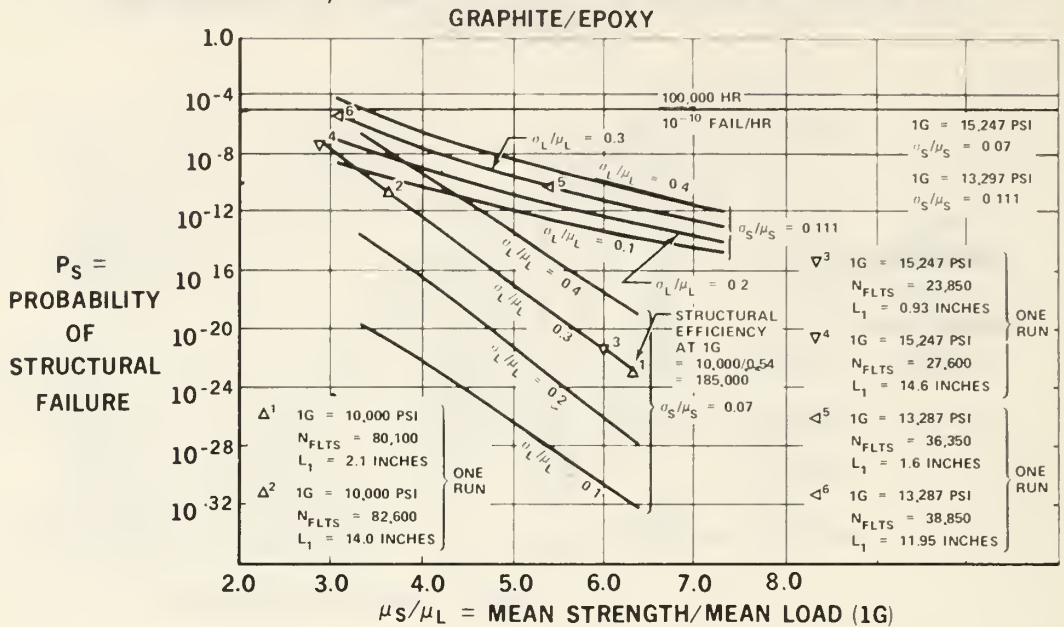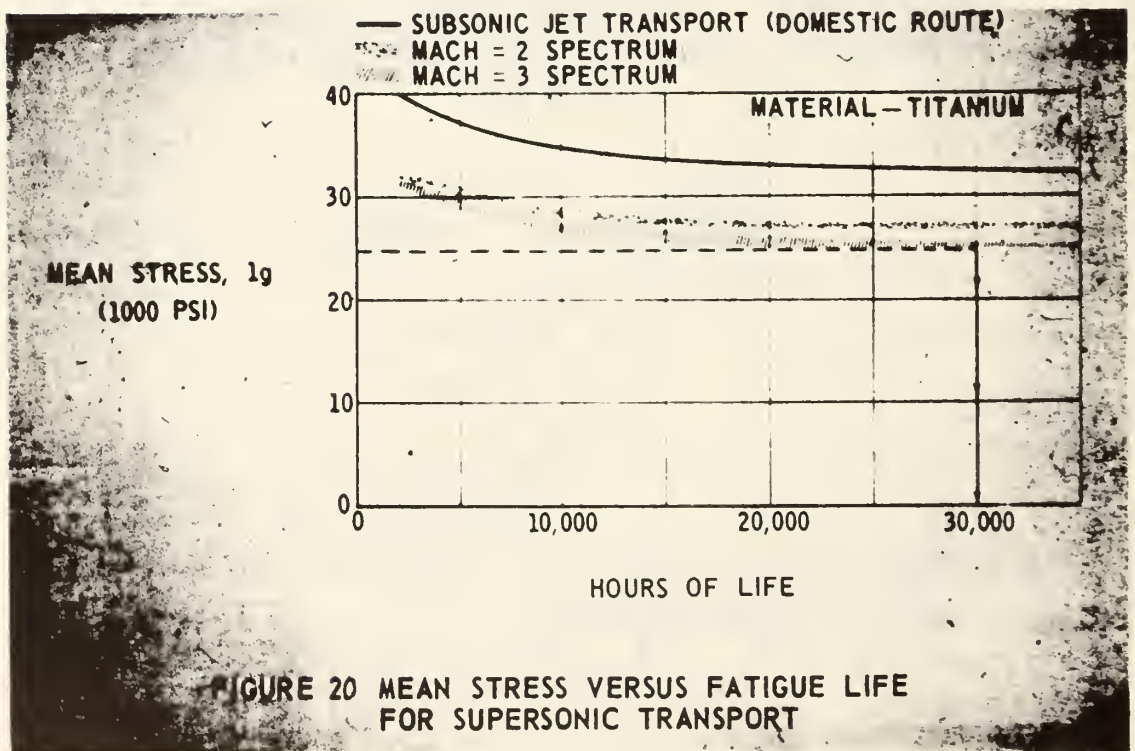
Fig. G-20

RADIAL LOAD DUE TO CABIN PRESSURE
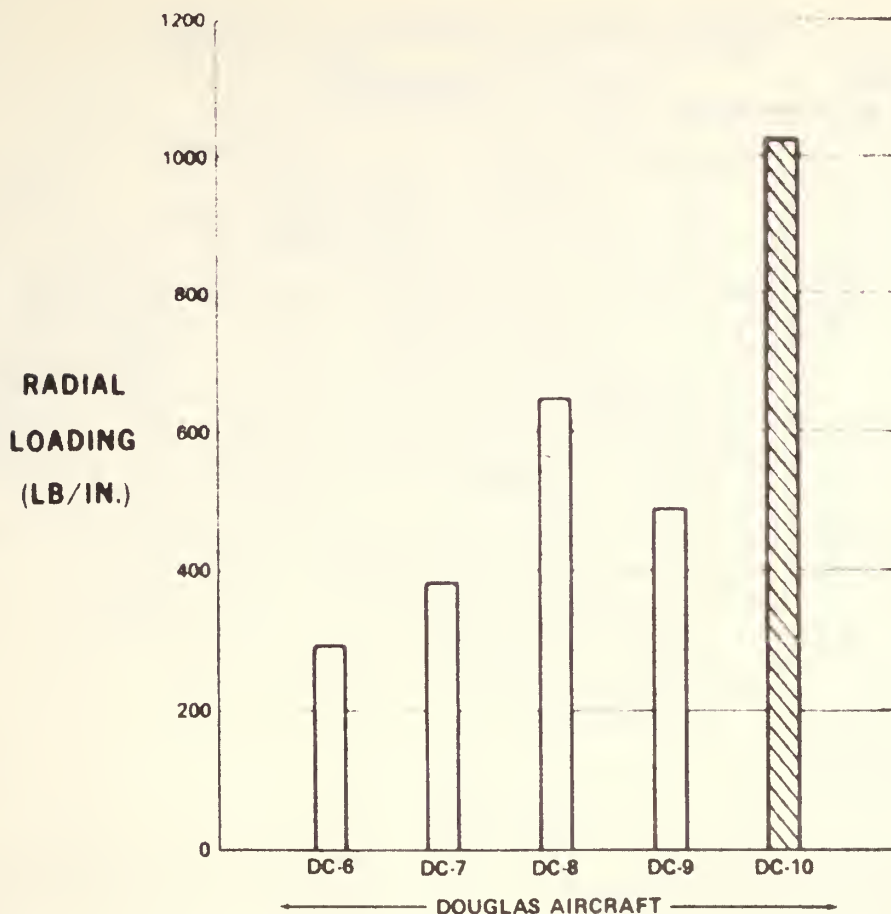
Fig. G-21

FUSELAGE FAIL-SAFE CRITERIA

- ALL FLIGHT AND PRESSURE STRUCTURE FAIL-SAFE

- TITANIUM CRACK STOPPERS (DC-8 TYPE)

- 2024-T3 USED FOR FUSELAGE SHELL AND LOWER WING

- RECOMMENDED INSPECTIONS BASED ON ANALYSIS AND TESTS

- LANDING GEAR, PYLON AND FLAP WILL BREAK AWAY WITHOUT TANK RUPTURE

DOUGLAS CRITERIA (100% LIMIT)

F.A.R. CRITERIA (80% LIMIT)

4/2/68

Fig. G-22

AXIAL LOAD
(INERTIA PLUS PRESSURE)

60 IN.

48 IN.

C

A

72 IN.

B

80 IN.

HOOP TENSION

100 IN.
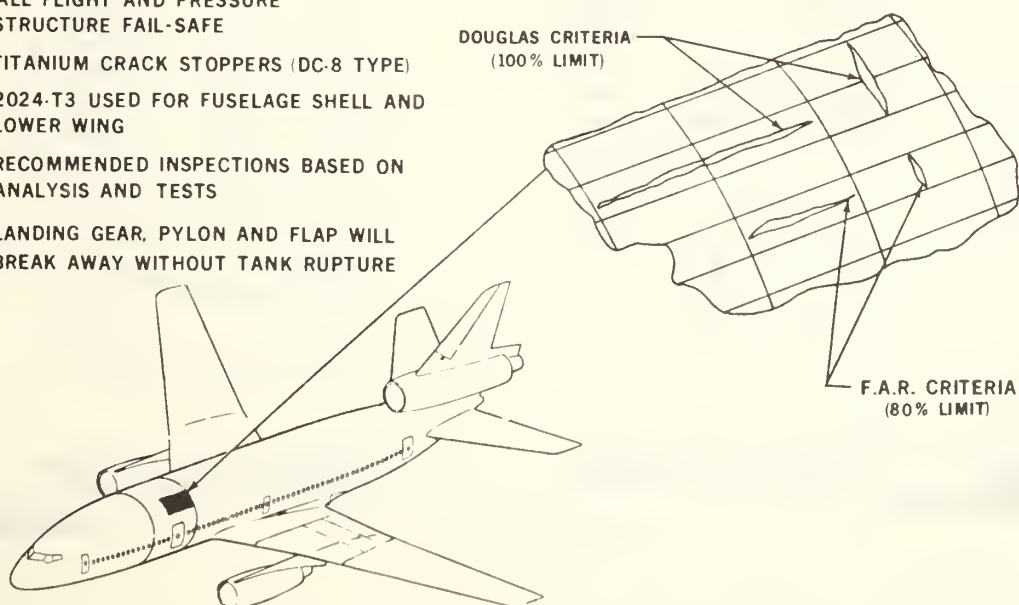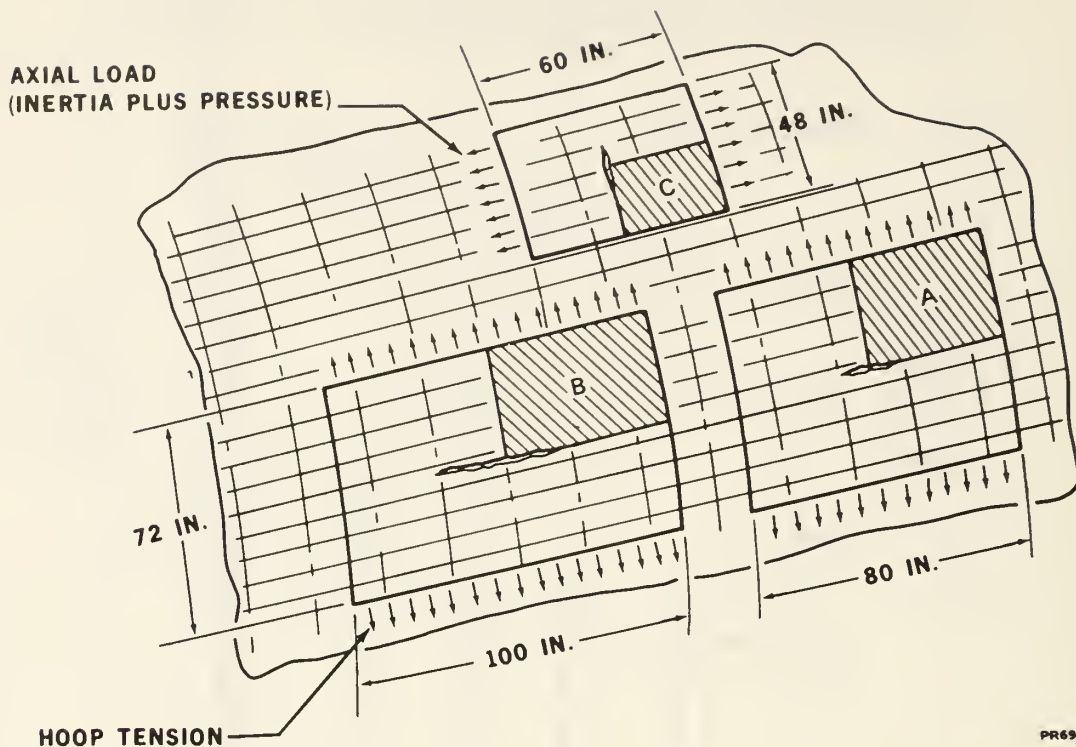
12/4/69

PR69-GEN-24294

Fig. G-23

COMPARISON OF MATERIALS FOR DAMAGE TOLERANCE
7075-T73 VERSUS 2024-T3

MATERIAL	K_C PSI√IN	GROSS RESIDUAL STRENGTH σ_R PSI
7075-T73	100942	29600
2024-T3	>168631	40850

MATERIAL	K_C PSI√IN	GROSS RESIDUAL STRENGTH σ_R (PSI)
7075-T73	65787	20000
2024-T3	88090	26800

σ_R

σ_R

σ_R

σ_R
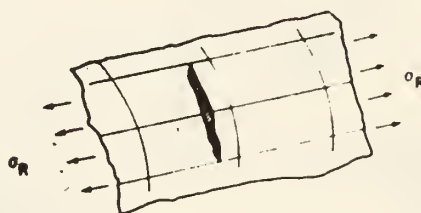
TRANSVERSE TWO BAY CRACK WITH
BROKEN CENTRAL LONGERON
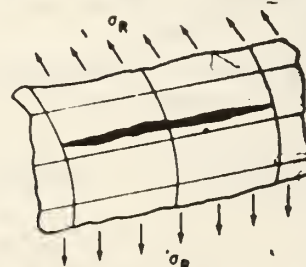
LONGITUDINAL TWO BAY CRACK WITH
BROKEN CENTRAL CRACK STOPPER

Fig. G-24

I-120

EFFECT OF THICKNESS ON STRESS
INTENSITY FACTOR

STRESS
INTENSITY
FACTOR

K_{Ic}

THICKNESS
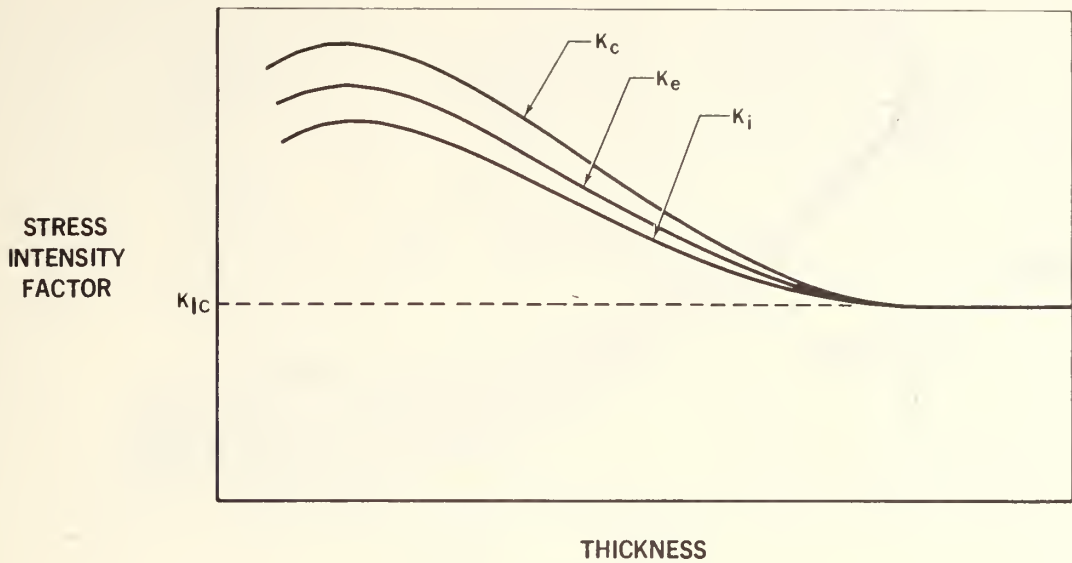
Fig. G-25

RESIDUAL STRENGTH OF UNREINFORCED PANEL

FAILURE STRESS VERSUS FINAL CRACK LENGTH

$\{ \ \overline{\underline{\text{I}}} 2a \quad \Rightarrow S$

FAILURE STRESS VERSUS INITIAL
CRACK LENGTH

S_C

STRESS

S_I

STRESS FOR ONSET OF GROWTH

$2a_0$ $2a_c$

CRACK LENGTH
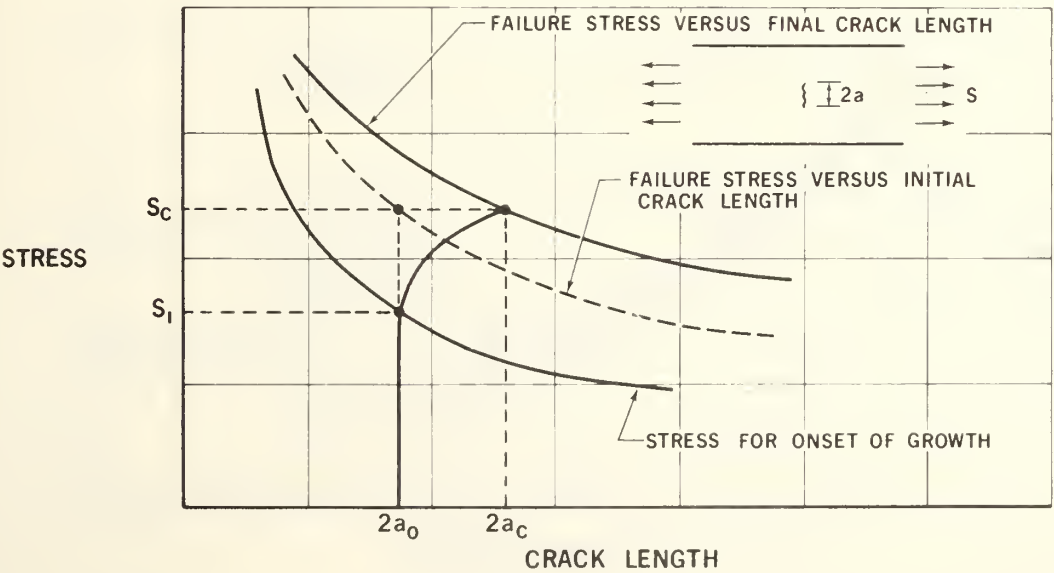
Fig. G-26

RESIDUAL STRENGTH CHARACTERISTICS
OF STIFFENED PANEL
STRINGER CRITICAL
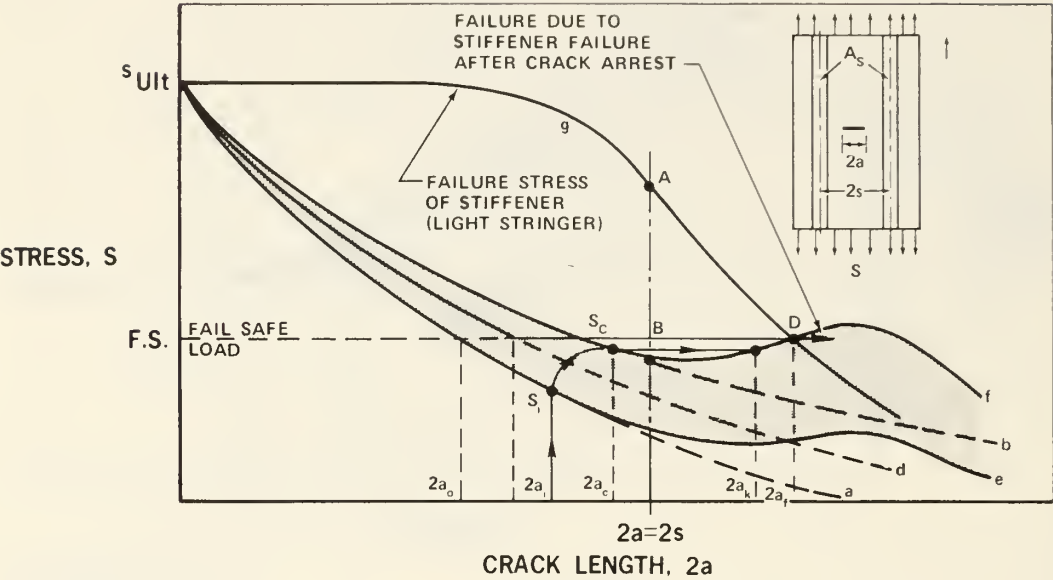
Fig. G-27

PR71-GEN-20540

RESIDUAL STRENGTH CHARACTERISTICS
OF STIFFENED PANEL
SKIN CRITICAL
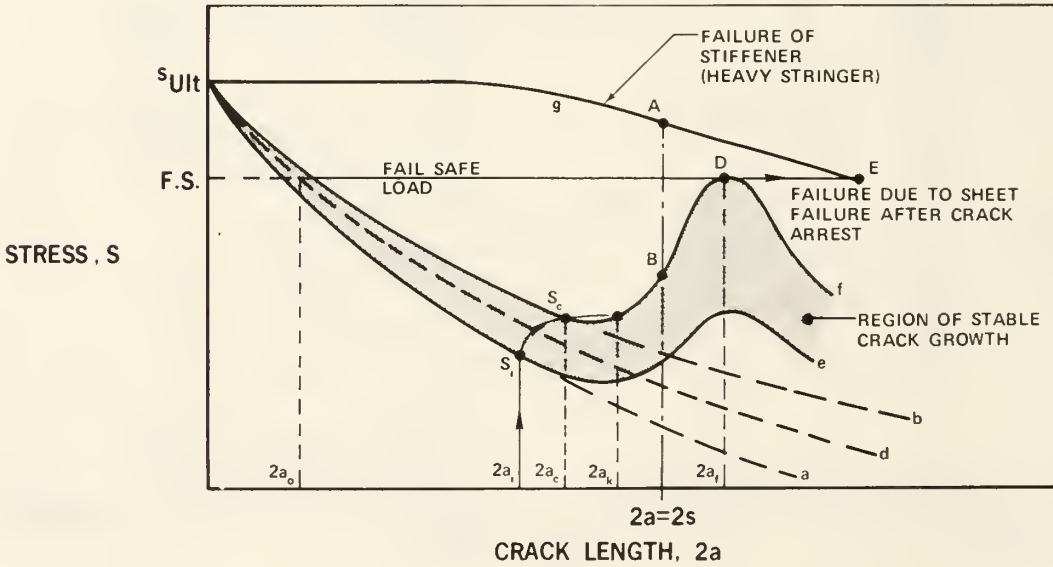
Fig. G-28

PR71-GEN-20534

FUSELAGE PANELS
(FAIL SAFTY)
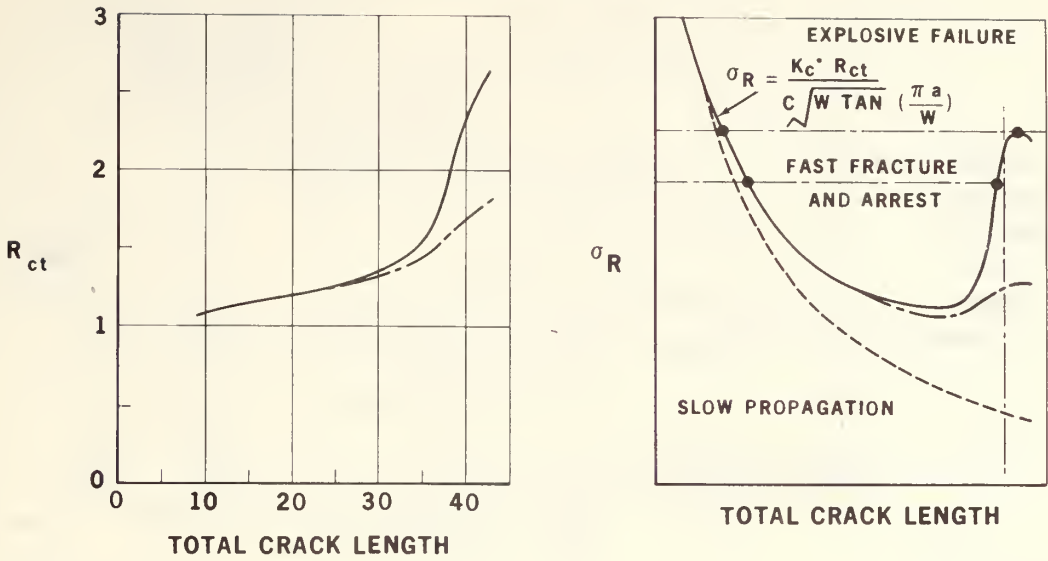
Fig. G-29

FATIGUE-CRACK PROPAGATION IN A PANEL REINFORCED WITH ALUMINUM-ALLOY STRINGERS SPACED AT 6 INCHES
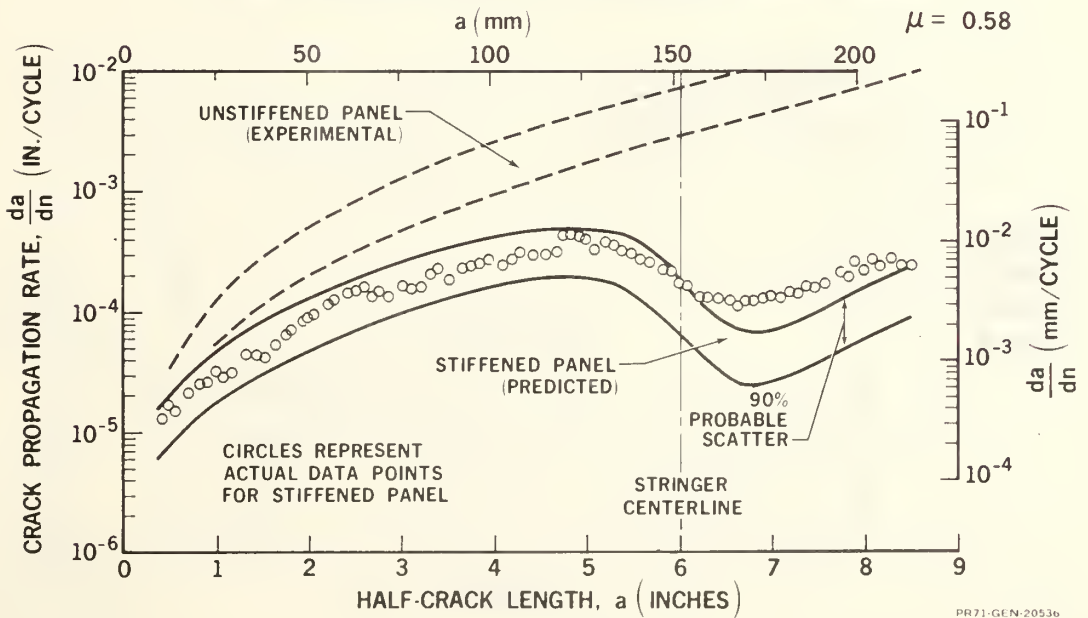
Fig. G-30

GROSS RESIDUAL STRENGTH CURVES FOR
CURVED PANEL NO. 15

Fig. G-31

PR69-GEN-24287

12/4/69
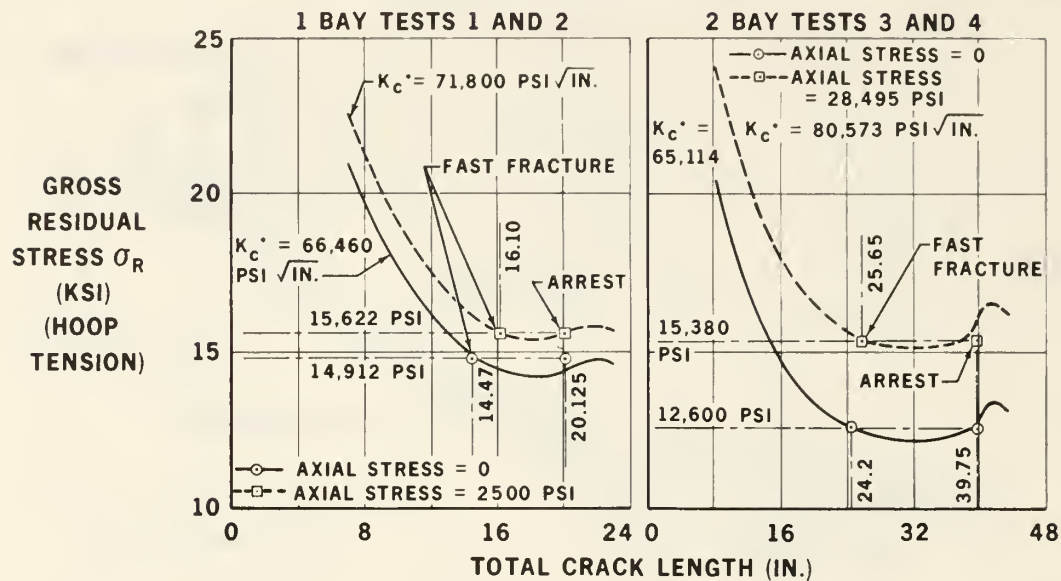
COMPARISON
OF THE FATIGUE CRACK GROWTH
RESULTS FOR TEST 2 OF PANEL 2 AND TEST 1 OF PANEL 4

12/4/69

PR69-GEN-24299

Fig. G-32

I-124
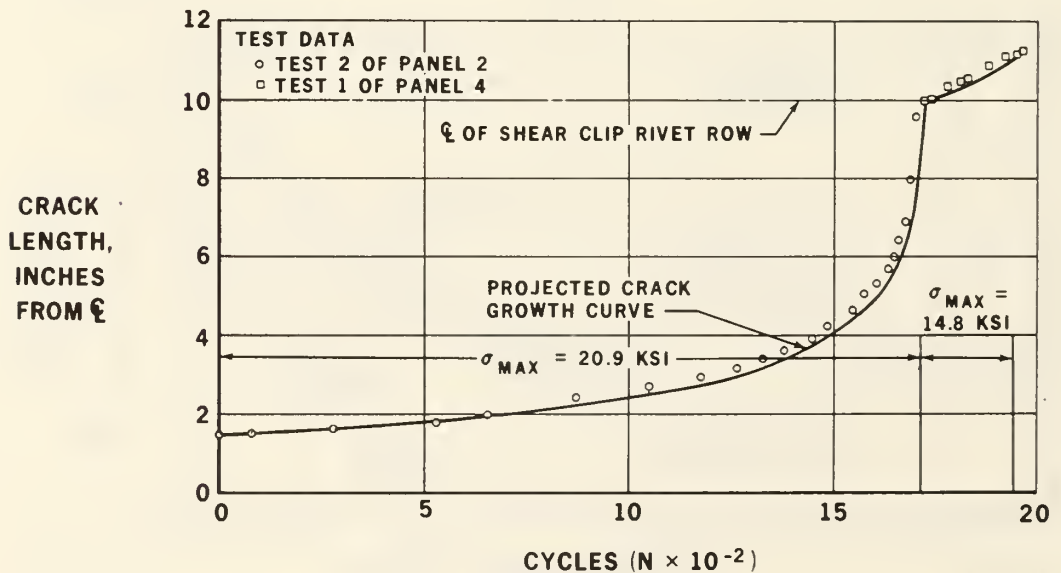
EFFECT OF OVERLOAD IN CONSTANT AMPLITUDE TEST
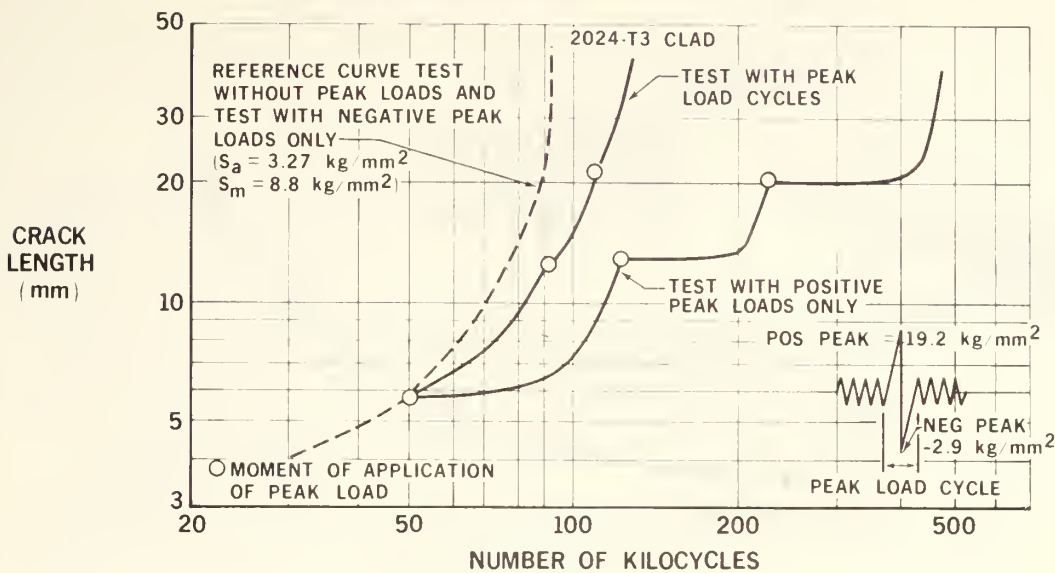
Fig. G-33

COMPARISON OF CRACK GROWTH RATE PARAMETERS WITH TEST RESULTS
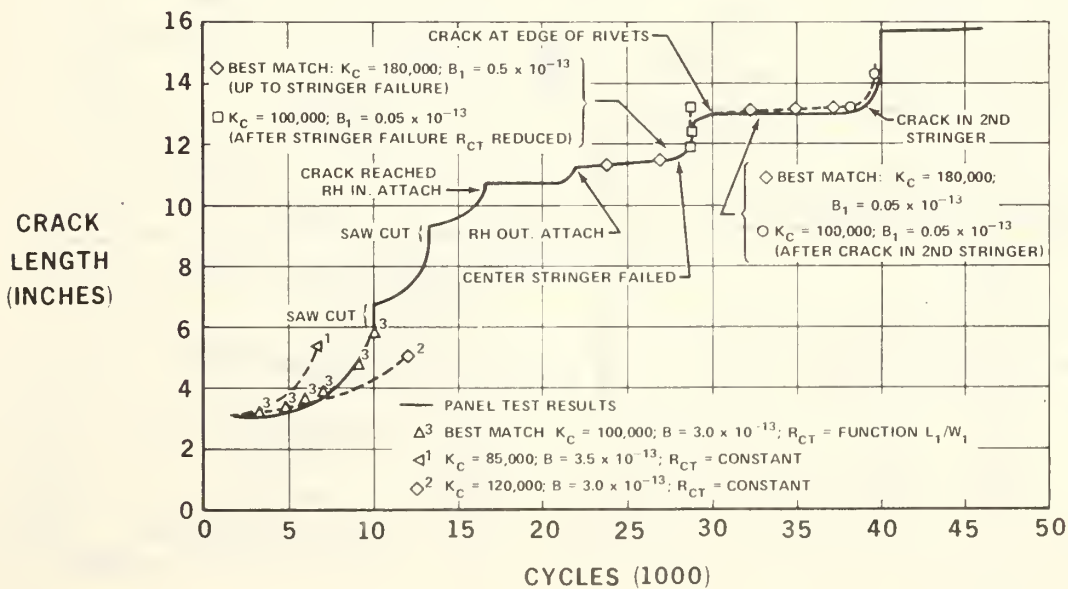
Fig. G-34

DELAY OF CRACK PROPAGATION AS A FUNCTION
OF PEAK LOAD MAGNITUDE
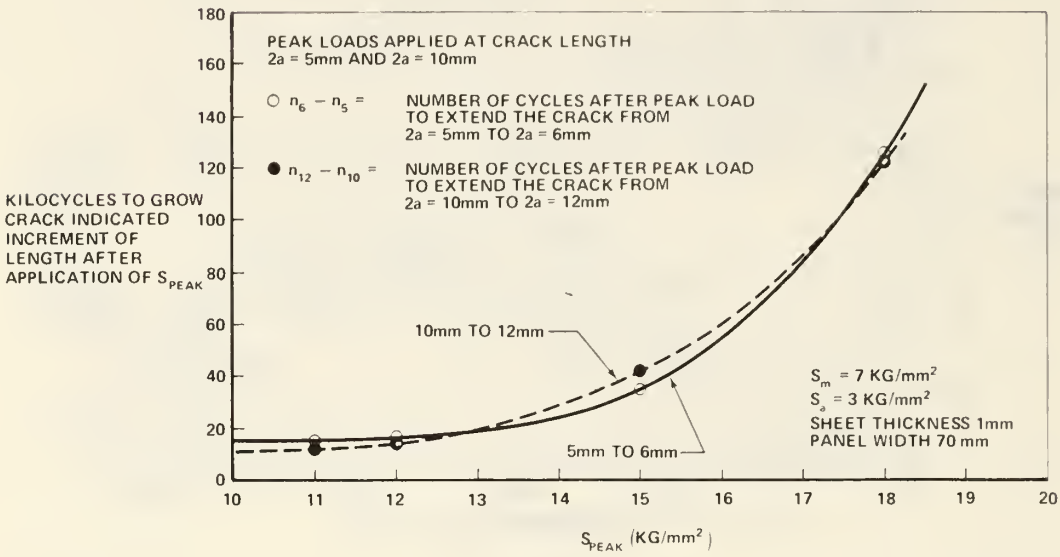
Fig. G-35
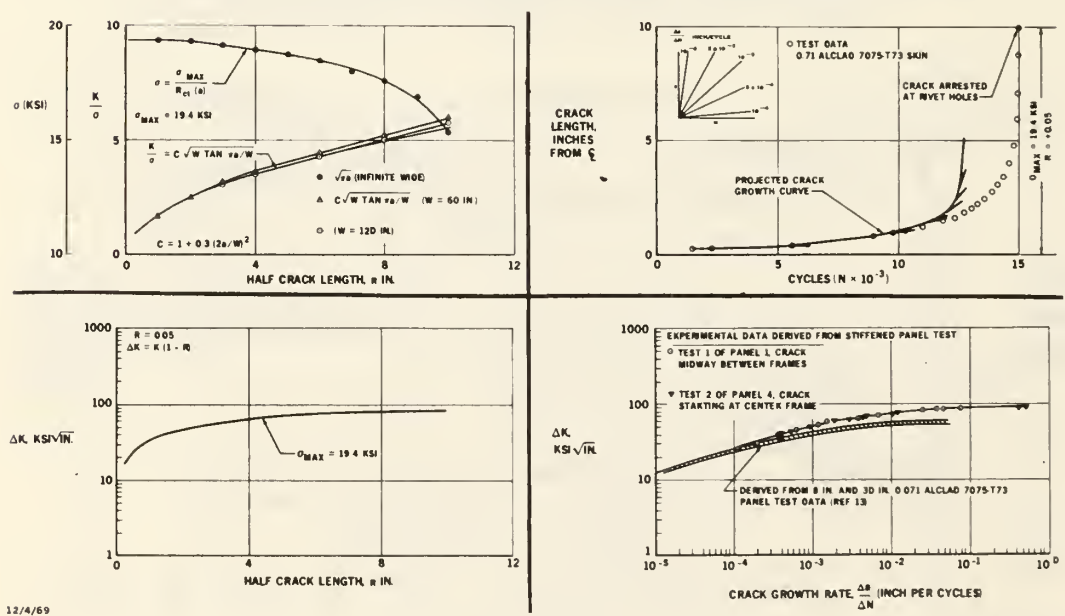
Fig. G-36

CONCLUSIONS

(1) PROBABILITY OF STRUCTURAL FAILURE GOOD TOOL FOR SELECTING MATERIALS AND TYPES OF CONSTRUCTION.

(2) ONE g STRESS MOST IMPORTANT

(3) SPECTRA VERY IMPORTANT

(4) STRENGTH VARIATION NEEDS MORE TEST DATA

(5) FRACTURE TOUGHNESS AND RATE OF GROWTH NEEDS MORE ANALYSIS AND TESTING USING FULL STRUCTURAL COMPONENTS.

(6) THE INSPECTION INTERVAL AND STRUCTURAL WEIGHT SAVING INFLUENCES THE:

    (a) RETURN ON INVESTMENT

    (b) D.O.C.

    (c) MAINTENANCE COSTS

(7) GRAPHITE/EPOXY WILL BE A GOOD CHOICE FOR WEIGHT SAVING AND STRUCTURAL INTEGRITY (LIFE) IF 1g STRESS LEVEL IS APPROXIMATELY 10,000 PSI.

Fig. G-37

PR71-GEN-20444

AVERAGE AIRFRAME COSTS
PER POUND OF STRUCTURE
1970 DOLLARS

The cost may be described by:

$\text{Cost} = (M + L) \times [\text{increments}]$

Where

M = Raw Material Cost $\times \dfrac{\text{Purchased Weight}}{\text{Flyaway Weight}}$

L = All Direct Labor Including Overheads

    $[\text{increments}] = (1+R) \times (1+E) \times (1+G) \times (1+P)$

R = increment for RDT&E

E = increment for ECP's

G = increment for G&A

P = increment for Profit

MATERIAL	RAW MATERIAL COST $/LB	FLYAWAY WT PURCHASED WT	LABOR COST	FACTORED TOTAL COST PER LB 200 UNITS	500 UNITS
Aluminum	0.66	0.359	105	148	107
Steel	0.18 to 1.00	0.049	113-140	196	166
Titanium	11.15	0.308	227	366	309
GFRP	5.31	0.862	165	237	171
Beryllium	360.00	0.308*	656	1750	1480

*Assumed Same as Titanium

PR71-GEN-20541

Fig. G-38

SPECIFIC STRENGTH vs SPECIFIC MODULUS
METALS vs COMPOSITES
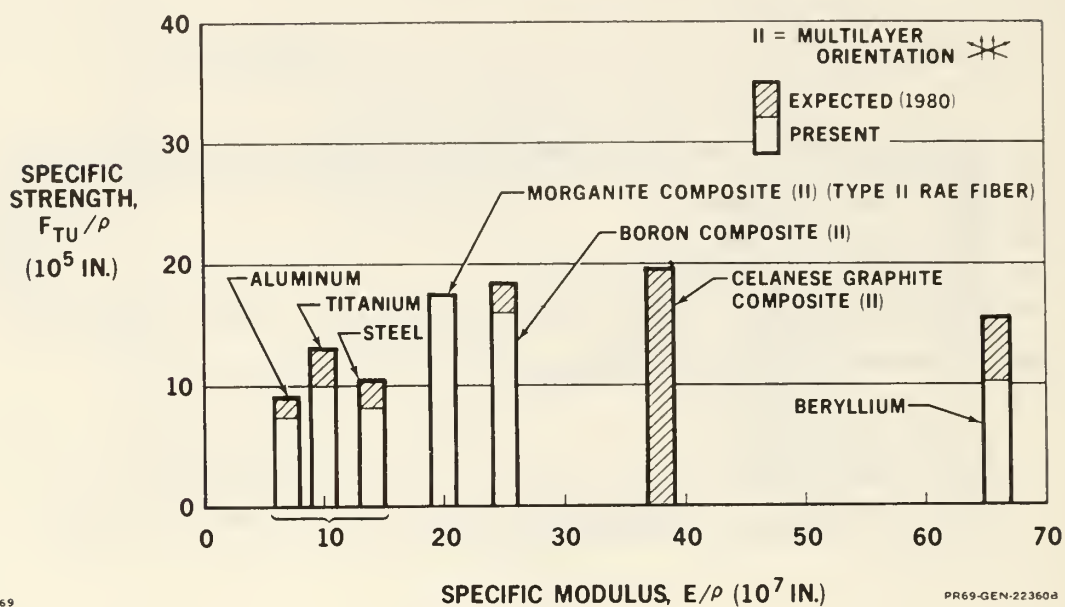
Fig. G-39

REDUCTION IN VEHICLE STRUCTURAL WEIGHT
BY USE OF HIGH MODULUS COMPOSITES
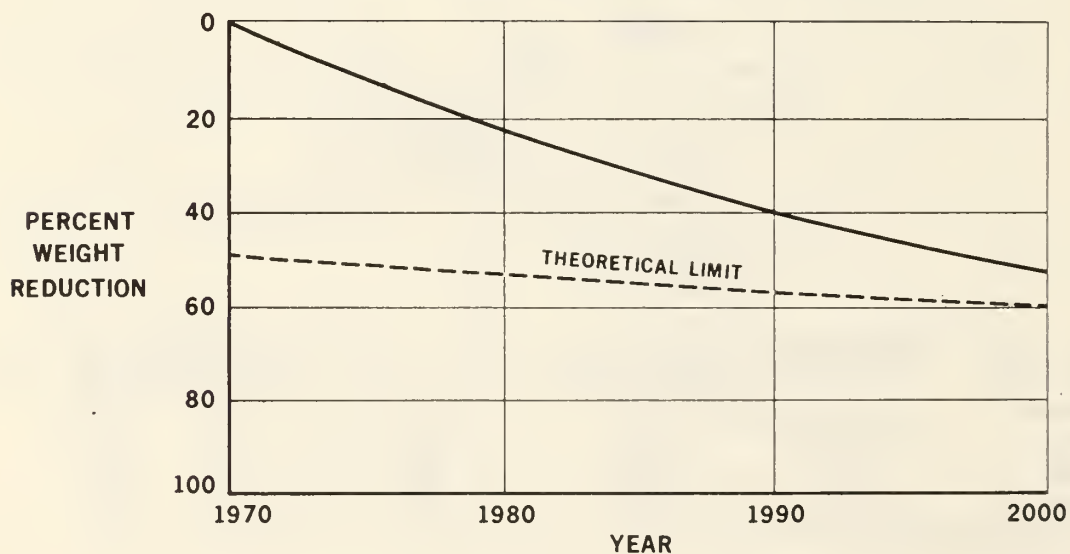
Fig. G-40

REDUCTION IN TAKEOFF WEIGHT
DUE TO 1985 TECHNOLOGICAL IMPROVEMENTS
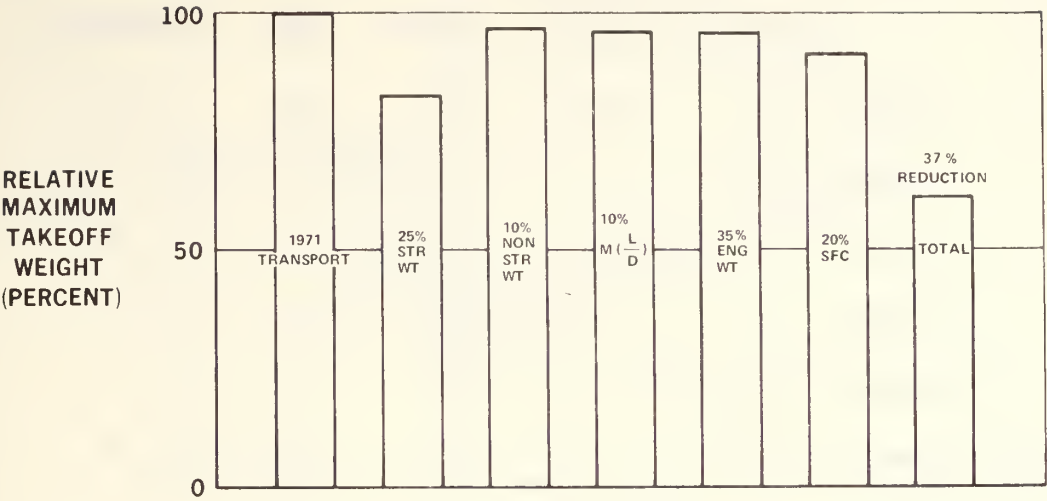
PR69-ADCS-1739C

Fig. G-41

REDUCTION IN DIRECT OPERATING COST
DUE TO 1985 TECHNOLOGICAL IMPROVEMENTS

8/5/69
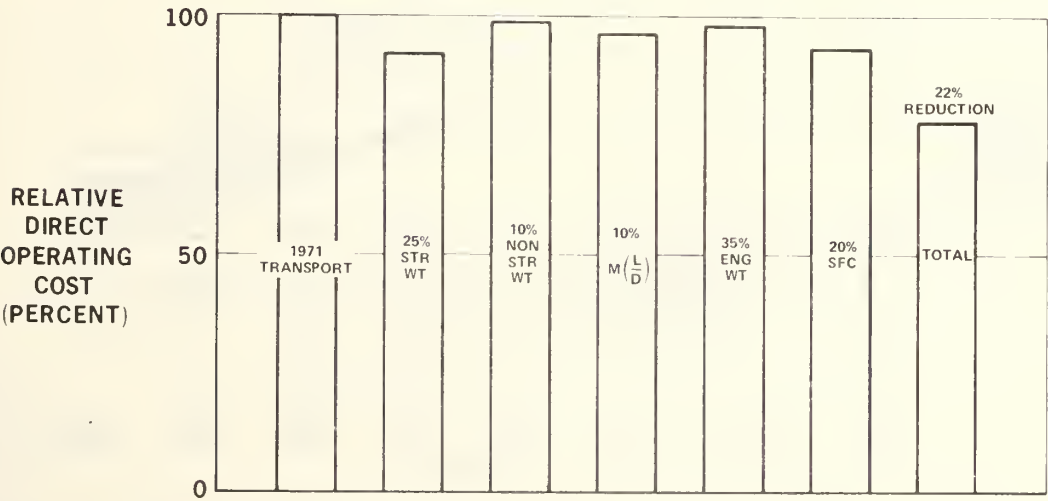
Fig. G-42

PR69-ADCS-1380C

TRENDS IN MATERIALS USAGE
PERCENT
(FOR PROPULSION)

	COMPOSITES	AI/MG	TITANIUM	STEELS	SUPERALLOYS
J47 (1950)	0%	22%	0%	70%	8%
J79 (1955)	0	3	2	85	10
J93 (1960)	0	1	7	24	68
GE4 (1965)	0	1	12	15	72
TF39 (1965)	2	1	32	18	47
NEXT GENERATION	5-10	1	25	15	50

J47 → J79: STEELS DEV; SUPERALLOYS DEV
J79: ALTERNATE DEV (TITANIUM ← 85); ALTERNATE DEV (→ SUPERALLOYS) 95
J79 → J93: TITANIUM DEV; SUPERALLOYS DEV; 92
J93 → GE4: TITANIUM DEV; 87
GE4 → TF39: TITANIUM DEV; 65
TF39 → NEXT GENERATION: COMPOSITES DEV; 65; 35

PR71-DC10-11229

Fig. G-43

SPECIFIC FUEL CONSUMPTION vs YEAR

MACH 0.8 CRUISE

SST

THRUST SPECIFIC FUEL CONSUMPTION LB/HR/LB

THEORETICAL LIMIT

THERMAL EFFICIENCY = 76%
PROPULSIVE EFFICIENCY = 100%

YEAR

'69 DC10-144478

Fig. G-44

I-130
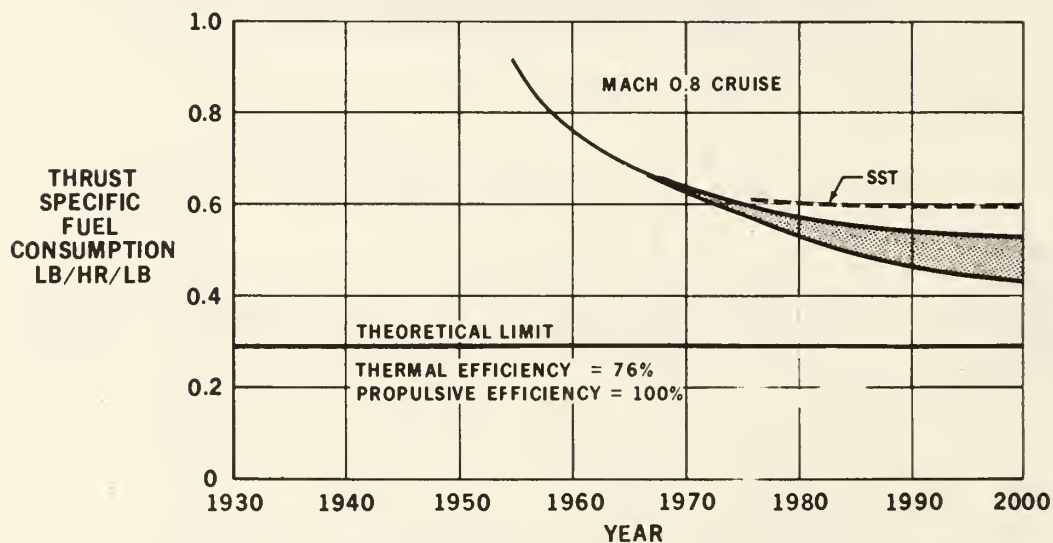
RANGE PARAMETER IMPROVEMENT
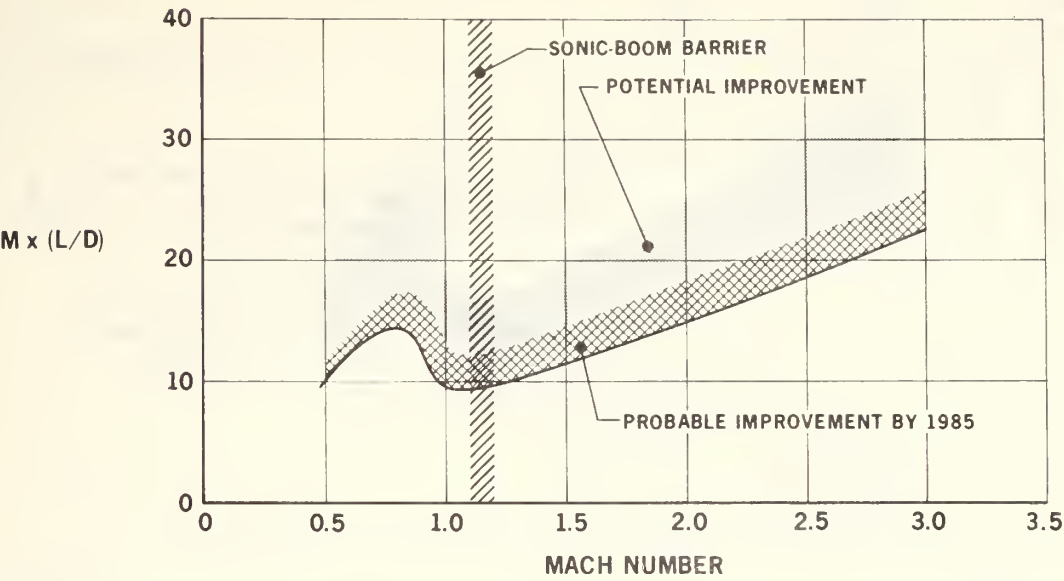
Fig. G-45

WEIGHT REDUCTION OF STRUCTURAL SYSTEM ITEMS
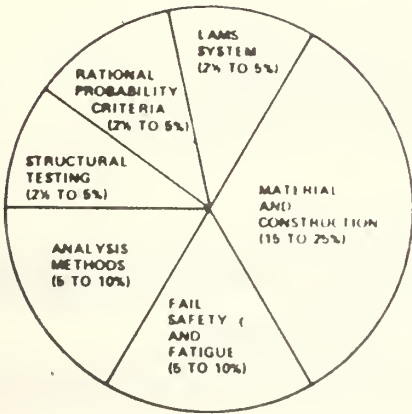
GOAL 45%
REDUCTION 33 TO 60%

Fig. G-46

DEVELOPMENT MILESTONES
STRUCTURAL AND DYNAMIC ANALYSIS

1950   1970   1990

—————— INTEGRATION OF DIRECT SYNTHESIS METHODS, DIGITIZED DRAWINGS, AND NUMERICALLY CONTROLLED MANUFACTURING

————— ADVANCED USE OF DIRECT SYNTHESIS METHODS

————— FULLY INTEGRATED ANALYSIS OF LOADS, STRENGTH, AND LIFE

———— DYNAMIC AEROELASTIC LOADS VIA INFLUENCE COEFFICIENTS

———— INITIAL USE OF DIRECT SYNTHESIS METHODS

———— ADVANCED COMPUTER ANALYSIS OF STRENGTH

——— PSD ANALYSIS OF GUST RESPONSE

——— INITIAL USE OF ELECTRONIC COMPUTERS (STRENGTH AND FLUTTER ANALYSIS)

8/4/69

Fig. G-47

PR69-GEN-22381

DEVELOPMENT TRENDS
STRUCTURAL AND DYNAMIC ANALYSIS

EMPHASIS
- AUTOMATION OF PROCEDURES
- INTEGRATION OF TECHNOLOGIES

BENEFITS
- ELEVATED VIEWPOINT FOR DESIGN ENGINEER
  - MORE EFFICIENT USE OF ENGINEERS
  - IMPROVED COMMUNICATION BETWEEN DISCIPLINES

- INCREASED RELIABILITY, SERVICEABILITY, AND EFFICIENCY OF STRUCTURAL SYSTEM

Fig. G-48

PLANS FOR FUTURE DEVELOPMENT

MATERIALS DATA

σ

E

LOFT DATA

EXTERNAL LOADS

P

% C

OPTIMIZATION PROGRAM

COMPUTER-DRAWN SECTIONS

APT

TAPE-CONTROLLED MACHINE
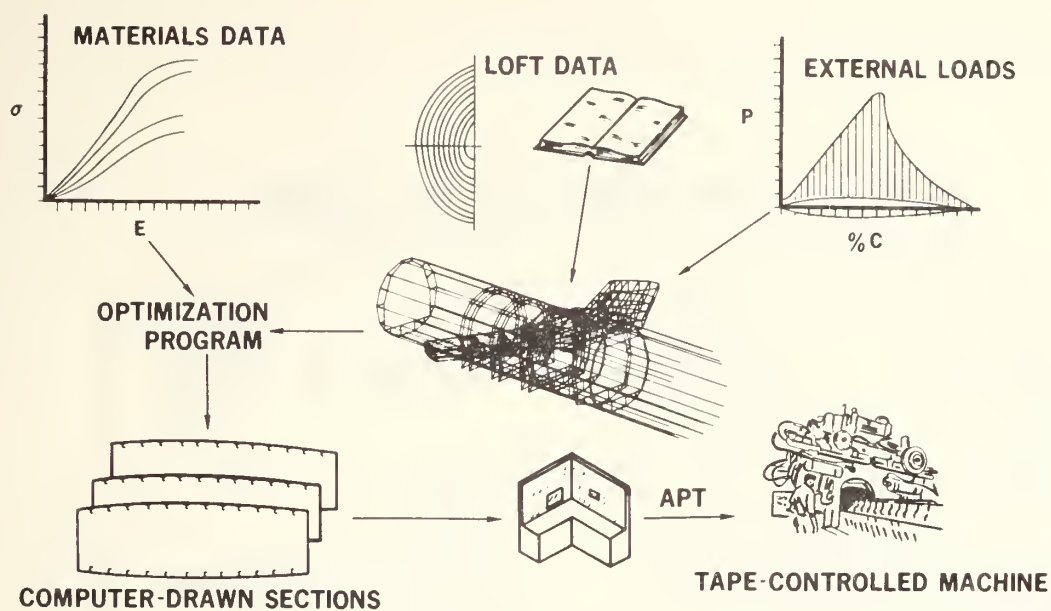
2/16/68

PR71-GEN-20459

Fig. G-49

STRUCTURAL DESIGN BY DIRECT SYNTHESIS
(FUTURE)

DESIGN CRITERIA

MATERIAL ALLOWABLES

STRUCTURAL ENVELOPE

BASIC AERO DATA

COMPUTERIZED DIRECT SYNTHESIS (OR AUTOMATED REDESIGN)

OPTIMUM STRUCTURE
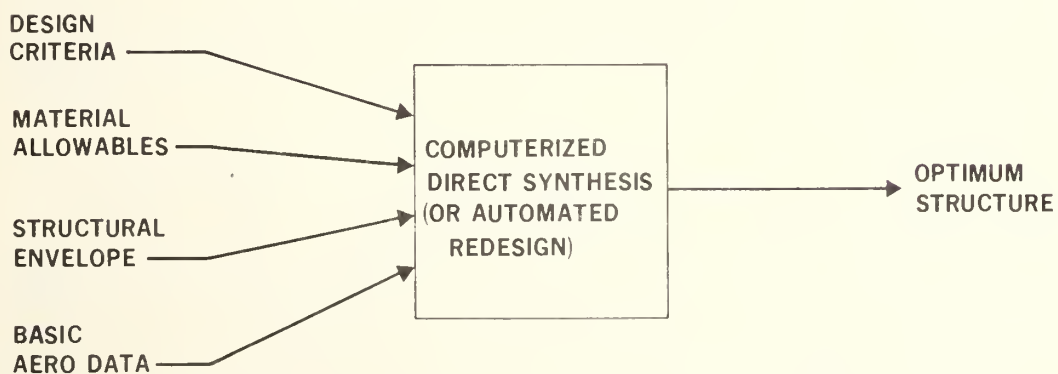
8/4/69

Fig. G-50

PR69-GEN-22382

PRELIMINARY STRUCTURAL DESIGN
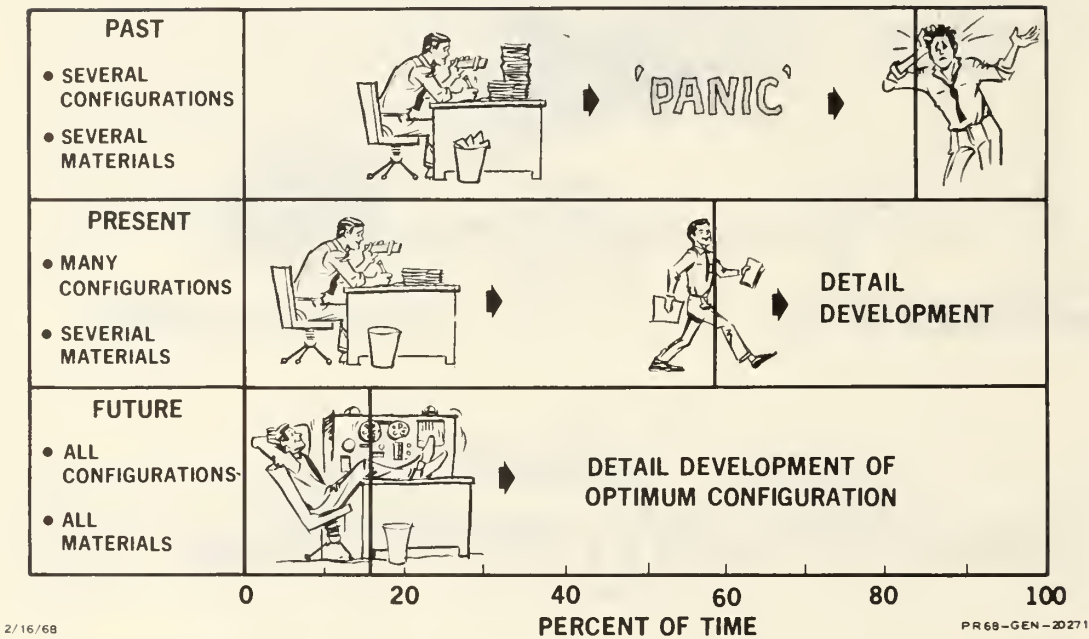
Fig. G-51

SOME PRACTICAL ASPECTS OF RISK EVALUATION

J. W. Ellis
North American Rockwell--Los Angeles

Abstract: Risk evaluation in aircraft design is still far from
being a useful tool. An approach is shown by
identifying uncertainty factors and impact factors
for determining a risk exposure index. Continuing
development is bound to take place in this field--
pointing toward an important tool in the design
decision process.

I almost feel that I should apologize for introducing such
an elusive subject as risk evaluation in a meeting encompassing
so many discussions on good solid, data-substantiated, hardware-
oriented problems. Why then, is this subject included here? Why
not leave this tenuous field to a management symposium? It is
true that, historically, risk has been primarily the province
of the manager, but the simple truth of the matter is -- the
manager needs help!

I am neither a program manager nor risk analyst, but in
my activity in preliminary design, whenever decision time comes
around, I feel a strong need for some methodology in the risk
area. To be of value in the detail decision procedure, this
methodology must, of necessity, produce quantitative information
which can be entered into the decision process much the same as
weight or cost. The goal of applying rational risk analysis
at the detail design decision level is not yet attainable, but
recognition of risk as a valid parameter in the decision process
has been made, and the first steps toward rational analysis
methods are being taken.

The Decision Process

Let us examine what is involved in the structural design
decision process. I believe everything that goes through the
mind of the decision-maker will fall into one of the four
categories shown in Figure H-1. The first one, the cost-weight
combination, of course, is a pretty large piece of our structural
world. It is considered here as a combination because I think
we have learned to deal with cost and weight as a trade. They
are not treated independently as they once were. Through value
engineering techniques which enable us to equate weight changes
with airframe growth, and subsequently with cost, we are able
to reconcile the weight and cost dichotomy. The second item,
reliability, is rapidly being quantified through developments
in fatigue analysis, fracture mechanics, fail-safe design
techniques and statistical approaches to structural failure

prediction. Thus this factor is approaching the point where it can be included in a quantitative way in the decision process. Schedules, of course, have always occupied a commanding spot in the decision process. We not only want to get our airplanes out on time, but once we get locked into a master schedule, we know we have to contribute a lot of blood, sweat and tears, and frequently money, to change. Design decisions dependent upon schedule factors are usually made on a go/no-go basis and thus require but little methodology to rationalize once scheduling is established.

The fourth decision factor is risk. It is a different type of consideration from the other three and is certainly the most elusive. Risk can be considered a quality of the cost/weight, reliability, and schedule factors in that it pertains to the likelihood of achieving goals in these areas, hence the bracket on the chart indicating that risk encompasses the other three. Certainly there can be no doubt that in spite of its elusiveness, it is essential to consider exposure to loss when we are about to make a design decision. Such consideration requires methodology, both to assess the risk, and to express it in quantitative terms compatible with the other factors under consideration.

Risk Evaluation

What is risk evaluation? It has been said that it is a sophisticated way of turning chicken and I think there is something to that; however, the subtle art of turning chicken at just the right time is one of the key requirements of remaining solvent in your job, your company, or your program. The manager who always takes minimum risk, goes down in flames when technology or his competition catches up with him, and the manager who disregards risk, follows the same path, but his flames have historically proven to be not only bigger but substantially hotter.

Let us examine the basic elements of risk evaluation as shown in Figure H-2. All risks involve both uncertainty and impact. There can be no risk without both of these items present. If you have a great uncertainty, but no impact as consequence of the uncertainty, then there is no risk; or conversely, if you have a great impact but you know with certainty how things are going to come out, there is no risk there either. So we have to look at both elements of this pair before we can say we have looked at risk.

Much of the imput data on the nature of the uncertainty will be subjective. Generally it must be learned from opinions, and but little will be clearly defined. However, this data must be quantified or we cannot work with it from an analytical standpoint. Risk impact may have a number of forms, but is usually resolved in terms of dollars, schedule impact, or system performance impact.

After we have evaluated the uncertainty and the impact, I
think a professional judgment will always be involved in arriving
at the final decision, at least in the foreseeable future.
I don't believe any of the risk people claim that we will have
an automatic system that circumvents or eliminates the judgment
factor. The analytical risk work is properly used to place
this judgment on a sound factual basis. The decision may be
based on one of several criteria. It may be a decision to
minimize risk as is shown in the figure, but it may more often
consist of a balancing of risk versus advantages.

Stumbling Blocks

There are a number of stumbling blocks which will always
render the job of risk analysis difficult. Several of these
are shown in Figure H-3. Uncertainty assessment most frequently
is a subjective matter especially when we are dealing with a new
subject where we have only limited historical data. It is
subjective because we must gather our working information by
asking people to express opinions on risk. We often will have
no other way to obtain information. The uncertainty judgment
required may be a complex one. Other uncertainties may be
involved, and it is difficult for the human mind to correlate
several subjective judgments.

In the area of impact assessment, there are often multiple
alternatives rather than a single valued solution. Impacts may
also be subject to a domino effect where one event will trigger
off a number of others requiring the ability to analyze the
system well enough to know the full extent of the dislocations
created.

Possibly the most formidable stumbling block is that of
arriving at a decision after the evaluating information is
obtained. Frequently, the evaluation may yield probabilities
of impacts in cost, schedule, and system performance simultaneously;
a mixture of apples, oranges, and grapes. A common denominator
is required to reduce the problem to manageable form.

Qualitative Risk Evaluation

Several approaches to risk evaluation exist. Figure H-4
presents a very simple one. I call it the qualitative approach.
It is based on an off-the-top-of-the-head probability evaluation,
quantified slightly by stating whether there is a high, medium
or low probability of failure. Some recognition of the impact
is taken by estimating, generally, what kind of trouble a
failure would bring. The final decision then is made by
invoking the powers of a "Big Daddy" in the organization. His
decision may be arbitrary, but it is very often unassailable.

Before we sneer too much at this type of approach, I think we should recognize that it has two things going for it; It is head and shoulders above no risk analysis at all, and it is the most commonly used method of all.

Risk Index Approach

A significant advancement over the approach just described is in use in some segments of the industry today as a comparative method of risk analysis. This is known as the risk index method, and is diagramed in Figure H-5. Here we introduce a systematic quantification of the risk factor. First we identify the risk areas and then break them down into uncertainty factors and impact factors. Next we identify the uncertainty factors involved, observe their relationships, and segregate them. This simplification makes it easier to think about the problem as we gather subjective data from our personnel. We don't try to treat more than one factor, or one coupling of factors at a time. Rather than thinking about the whole problem, we look at one piece at a time and assign probability indices to each piece. These are usually expressed as percentage figures because we are used to thinking in percentages.

Similarly, we look at the impacts, which may be in terms of cost, schedule, or performance and assign an index to them. Frequently this index can be reduced to the form of dollars. Finally a risk exposure index is derived by multiplying the uncertainty, or probability of failure index, by the impact index. Because of the somewhat arbitrary nature of the index factors, this method is most commonly used to compare the risk exposures of alternate solutions to a design problem. When used in this manner, this technique can be remarkably effective at the detail design level. Although useful as a means of comparison, this method is more of a straightening-out-of-thinking than it is real methodology.

Figures H-6 and H-7 show an example for a real design problem. In this instance, consideration is being given to a design change to composite construction for an aircraft component, at a good saving in weight. Risks are involved, both in the basic material properties to be used in design, and in the validity of composite component analysis methods. In the first case, any error would be detected early in the program through element tests, while errors in analysis would become apparent only late in the program in component testing. Assigning subjective estimates of probability of failure and applying the risk index analysis results in a risk exposure index of $55,000 as shown in Figure H-7.

The index may be interpreted in several ways. It may be
compared with similar analyses of alternate approaches to
achieving the same goals, or it may be treated in a probabilistic
manner by declaring that any means of reducing the risk to zero,
such as early tests, etc., which can be accomplished for
$55,000 or less, is a good action to take. Another evaluation
may be made by comparing the risk exposure index to the dollar
value of the weight saved as determined through value engineering
analysis.

This is about the present limit to the application of risk
analysis at the detail design decision level. The method has
several basic deficiencies which limit its value. It does not
consider a range of probabilities of failure versus various
degrees of impact, only discrete points on the curve. It also
cannot deal with the interaction of a number of probabilities
and consequences. In addition there is no provision for converting
the direct consequences of a failure to meet a goal into a
relevant impact upon the system. Finally, there is no attempt to
provide a valid comparison between the risk exposure and the
advantages to be gained. It appears that a more sophisticated
methodology is needed.

Rational Probability Risk Analysis

There has been some work done in the associated field of
contract management and incentive fee allocation, employing a
rational probability approach which may answer some of the
objections stated above. In Figure H-8, I have diagrammed a
hypothetical approach to risk evaluation using some of the
elements of rational probability analysis and, in addition,
answering some of the other objections stated above. Here we
take the proposed concept and identify the risk areas as before.
Then we set up a probability network which relates the various
facets of uncertainty involved in the problem. We relate them
and assign values through a procedure similar to the one shown
for the risk index analysis, except that, instead of discrete
values for probabilities and consequences we describe them through
probability curves relating probability of occurence with
consequences over a range of values. Through the probability
network analysis we arrive at an output which is a resultant
probability versus consequences, again in the form of curve
data rather than discrete points. However, we are probably not
looking for the consequences as such, but rather for the response
of the system to these consequences. So we do a sensitivity
analysis to see what significance weight, delay, or any other
result really has to the system. From this we derive a relevant
impact, usually in the form of a cost, schedule, or performance
factors. We would like to reduce this to a common denominator
if possible, usually in terms of dollars.

We can next take the probability-impact data, put them into the form of a probability density function, and multiply probability times the impact as we did before for the risk index analysis. This will give us a peaked curve from which our maximum risk exposure is determined.

The next step is to compare this risk exposure with the concept advantage. In order to reduce the advantages to a comparable base, they are defined in terms of system advantages through sensitivity analysis, and then reduced to a dollar basis, much as the impacts were treated. When all of this is accomplished, we are then able to compare the risk and advantage on a dollars versus dollars basis.

The preceding example of an advanced risk analysis procedure was highly hypothetical, presented primarily to illustrate some of the problems which must be overcome in future development.

Problem Areas

One of the basic problem areas in risk analysis mentioned previously is that of subjective data collection. Figure H-9 illustrates the form of the probability data which is sought from experts in the technical areas. On the left side is shown a cumulative distribution curve. It is fairly easy for people to think in terms of what is least and most, or best and worst. This type of information is obtained first and forms the first points on the curves. The subjects must then be led to estimate at some other level, estimating 80% or 20% probability for instance; there are methods of interrogation which are being developed in the behavioral sciences, involving such things as choice of gambles and bets, to try to coax this kind of information out of people.

In addition to giving us these probability estimates, technical groups also have to tell us what the penalties for failure are. In this case, if we give them the cost, schedule, and performance variations, they can tell us what this means to the contract in dollars or to the reduction in system cost effectiveness. This data is generally derived analytically and is not subjective in nature.

The output of the network probability analysis can also be represented by a cumulative distribution curve, similar to the inputs in form. These distribution curves, as shown in Figure H-9, can tell us a number of things in themselves. For instance, the left-hand figure tells us that, considering all facts, all impacts, we have a 75% chance of meeting a certain performance level. Now, by applying the sensitivity conversion parameters for the system we can produce the right-hand figure. This tells us, for instance, that we have a 90% probability that the loss will not be greater than a certain amount.

Some other non-engineering problem areas are outlined in Figure H-10. The determination of overall total impact is frequently difficult because of factors not amenable to engineering analysis. The company reputation might suffer damage beyond the actual accountable dollars and cents if a project fails or is delayed by the failure of a risk item. Delays may cause damaging interference with other projects and failures or schedule slippage could conceivably jeopardize other contracts.

Most of our contracts have some kind of fee or penalty feature which can be included in the risk analysis. However, if we are striving to reach a wise and prudent decision through our risk evaluation, we may find that the contractural stipulations bias the decision unduly. Then there is the matter of self interest. Here again, the solution should be worked out on a mutual-basis with the customer. Unfortunately, very often the advantages you are trying to achieve accrue mainly to the customer and the penalties are all yours.

Management acceptance can be nearly insurmountable obstacle in some organizations. Among managers there seem to be two camps. There are those who agree that they most certainly need help in making decision involving risk. They recognize the value of quantitative help from the engineers who are most familiar with the system. Those in the other camp oppose any infringement upon their traditional ground by technical personnel. The solution undoubtedly will lie in development of viable risk evaluation methodology and the education of management in its use.

In conclusion, I believe it is safe to say that in spite of the visible problems, we will assuredly see continued development in the methodology of risk evaluation and that it will eventually alter the nature of the process by which we make our structural decisions. If properly applied, it should lead us around a lot of the blunders that we have perpetrated in the past in the application of new materials and constructions.

DESIGN DECISION

1. COST - WEIGHT

2. RELIABILITY   4. RISK

3. SCHEDULE

Fig. H-1

RISK EVALUATION

UNCERTAINTY
• SUBJECTIVE
• QUANTITATIVE

PROFESSIONAL
JUDGEMENT

RISK REDUCTION
DECISION

IMPACT
• $
• SCHED
• PERF

Fig. H-2
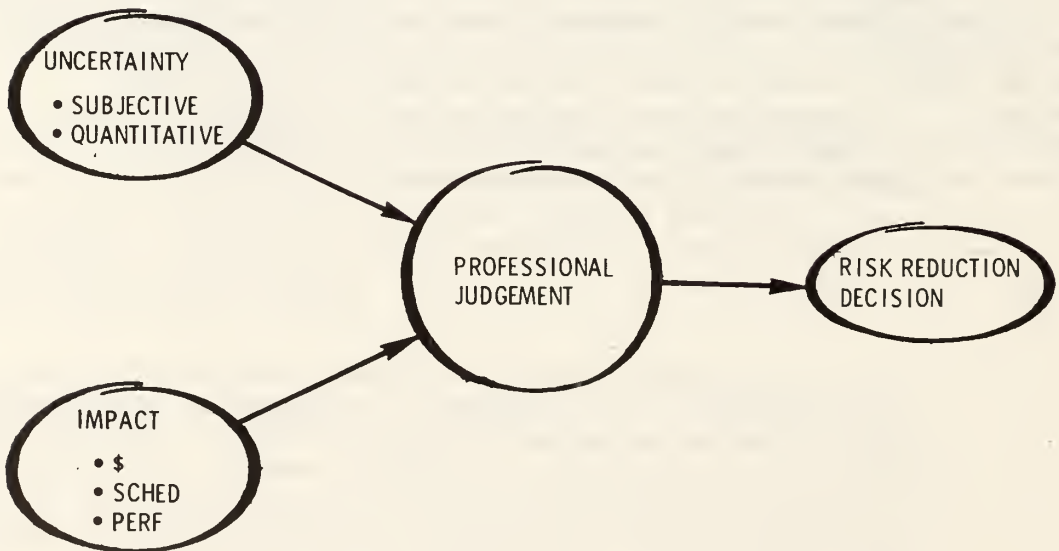
STUMBLING BLOCKS

- UNCERTAINTY ASSESSMENT

    - SUBJECTIVE
    - LIMITED HISTORICAL DATA
    - COMPLEXITY

- IMPACT ASSESSMENT

    - MULTIPLE ALTERNATIVES
    - DOMINO EFFECT
    - SYSTEM SENSITIVITY

- DECISION

    - APPLES, ORANGES, AND GRAPES

Fig. H-3

QUALITATIVE APPROACH

- TOP-OF-HEAD PROBABILITY EVALUATION - HIGH, LOW, MEDIUM

- SOME RECOGNITION OF IMPACT

- ARBITRARY DECISION BY 'BIG DADDY'

MOST COMMONLY USED

Fig. H-4

RISK INDEX APPROACH
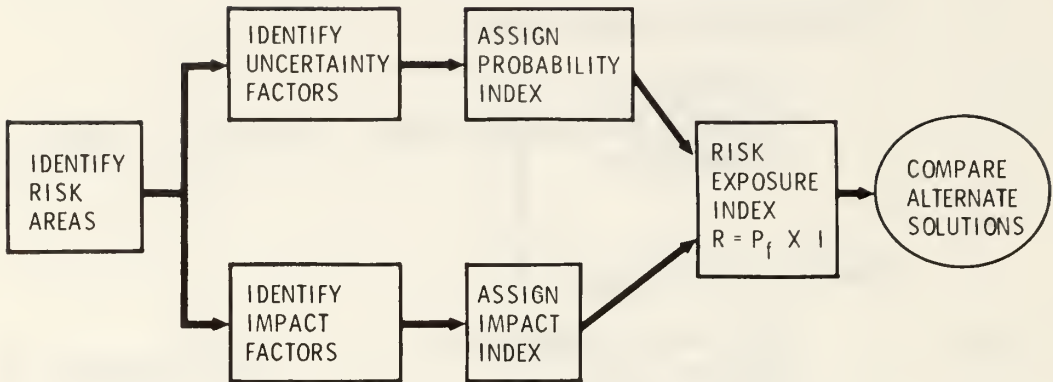
Fig. H-5

RISK INDEX ANALYSIS

QUESTION: SHOULD THE VERTICAL STABILIZER BE CHANGED TO GRAPHITE COMPOSITE CONSTRUCTION.

ADVANTAGE: GOOD WEIGHT SAVING AT REASONABLE FABRICATION COST

RISK AREAS: 1. BASIC MATERIAL CHARACTERISTICS

        UNCERTAINTY: COMPRESSION AND SHEAR ALLOWABLES USED IN DESIGN MAY NOT BE VALID

        IMPACT: REDESIGN OF ASSEMBLY AFTER ELEMENT TESTS

    2. COMPONENT PERFORMANCE

        UNCERTAINTY: VALIDITY OF GENERAL STABILITY ANALYSIS

        IMPACT: REDESIGN, TOOLING SCRAPPAGE, AND SCHEDULE SLIPPAGE AFTER COMPONENT TESTS

Fig. H-6

RISK INDEX ANALYSIS

RISK AREA	PROBABILITY (P_f)	IMPACT (I)		RISK EXPOSURE $R = P_f \times I$
MATERIAL PROPERTIES INADEQUATE	0.30	REDESIGN	$ 50,000	$15,000
COMPONENT PERFORMANCE INADEQUATE	0.10	REDESIGN TOOLING SCHED SLIP (O'TIME)	100,000 225,000 75,000 $400,000	40,000
		TOTAL RISK EXPOSURE		$55,000

Fig. H-7

RATIONAL PROBABILITY APPROACH
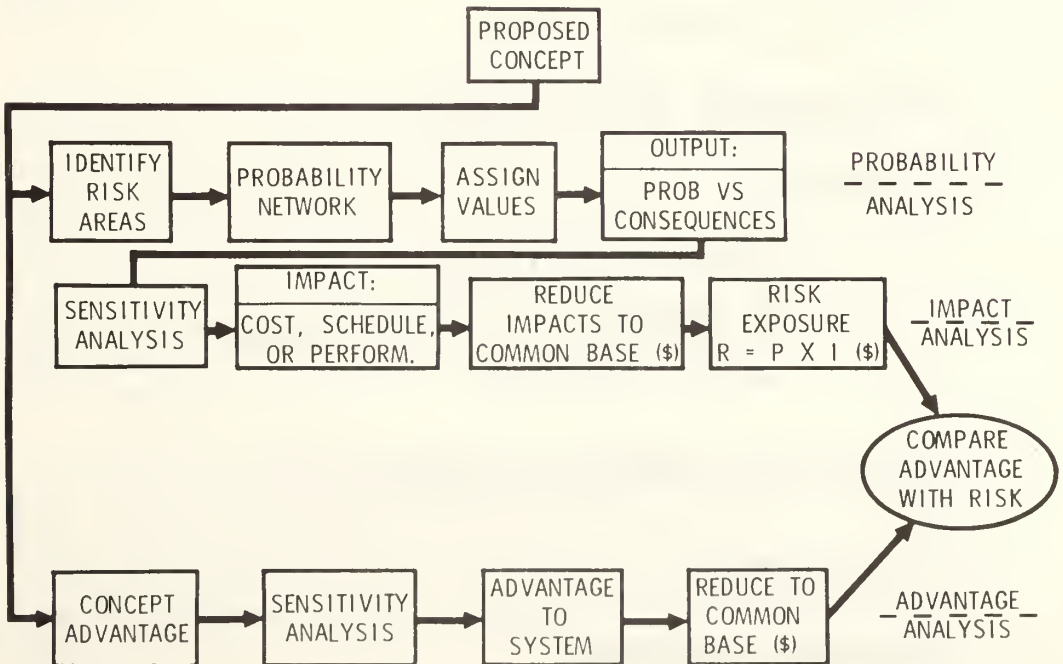
Fig. H-8

PROBABILITY ANALYSIS
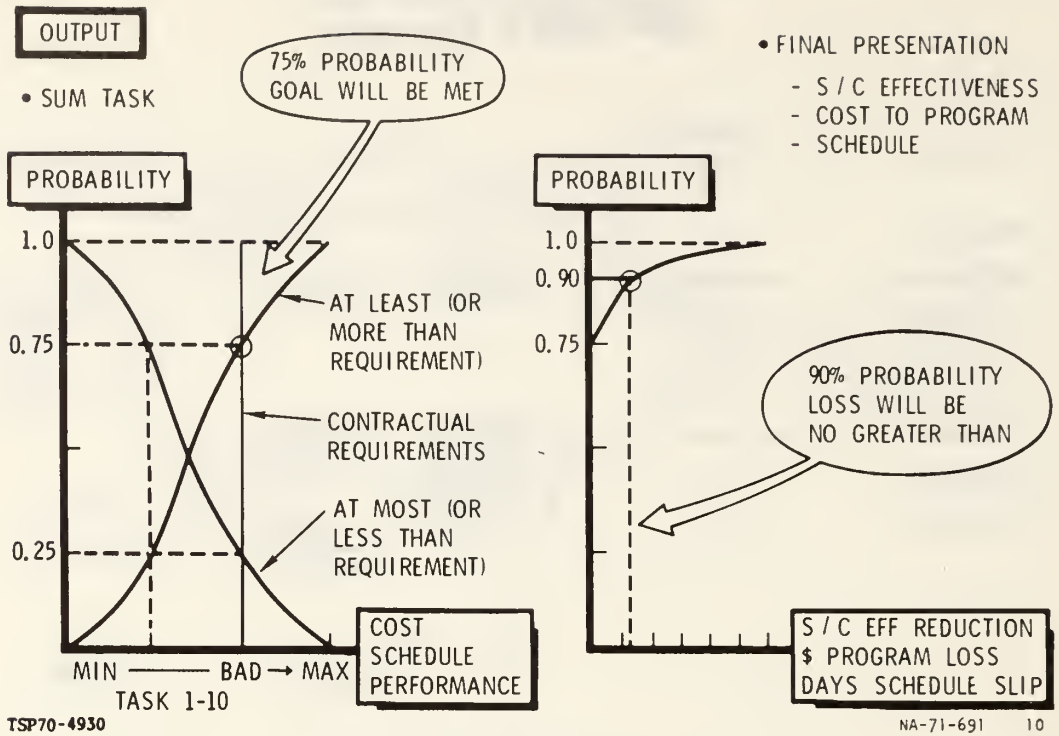
Fig. H-9

PROBLEM AREAS

- SUBJECTIVE NATURE OF DATA

- DETERMINATION OF TOTAL IMPACT

    EXAMPLES:

        - COMPANY REPUTATION

        - INTERFERENCE WITH OTHER ACTIVITIES

        - RELATED CONTRACTS

- CONFLICTING CONTRACTUAL FEATURES

- SELF INTERESTS

- MANAGEMENT ACCEPTANCE

Fig. H-10

ADVANCED METALLIC STRUCTURES

D. A. Shinn
Air Force Materials Laboratory

Abstract: The Air Force program on Advanced Metallic Structures
has the objective to increase structural efficiency,
integrity and reliability by sponsoring a systematic
approach to critical structural problems. The
technical requirements of this program are discussed--
emphasizing the dual aspects of applying technology
and disseminating the resulting information to the
technical community.

During the preceding presentations we have been listening
to many aspects of design problems. I would now like to give
you the outline of an approach toward their solution. Precisely
how these problems can be solved is, of course, up to the air-
craft industry. Our approach is to contract to the best offer to
do the job.

An Advanced Development Program has been initiated by the
Air Force Systems Command in the field of Advanced Metallic
Structures. Its objective is to demonstrate more reliable and
improved structural design methods, materials, fabrication
processes, and evaluation techniques. The underlying reasoning
can be found in some general trends which have become clearly
visible.

One consideration is that developments in structural
design have in the past been spearheaded by challenging tasks.
Traditionally, technological challenges have been posed by the
development of numerous new fighter and attack aircraft which
provided a proving ground for the designer's ingenuity. In
the early 1950's, an average of three new projects for fighter
or attack aircraft were started each year. During the last
decade and a half, however, this average has been less than one
in three years, i.e. a shrinkage by an order of magnitude,
permitting much less opportunity for new and varied structural
developments.

Another consideration is that confidence in new structural
designs can be established only by structural testing. Here
the lessons are learned for future developments. The record
shows that for more than 70 systems, tested over a long time
period, more than half had two or more major components failing
below design load. The percentage has not improved but actually
worsened for recent systems, indicating perhaps more rigorous

test conditions or, more likely, a basic need to improve analytical methods. A very small flaw can have drastic effects and cause the loss of an aircraft. We need to improve structural efficiency and to avoid structural failures but the lack of opportunity to demonstrate new structures has hindered technological developments.

We can see this in its proper context if we realize how the development of structures depends on basic research which feeds into exploratory development, then into advanced development, engineering development, and finally into manufacturing technology. These all interact with each other. They all provide increased confidence in design, manufacturing, and integration. Our current problem arises from the fact that nothing has been done in the advanced development area of structures. Every one has been preoccupied with their own specialties, with no forcing function such as an ADP to make them communicate and learn from each other.

The preceding discussion is an elaboration of the objective in the Advanced Development Program. Its purpose, as stated before, is fairly straightforward. The approach is actually to design and fabricate structures with best available technology and to test them as a check on analytical methods. Feedback to improve both analysis and test will be emphasized. Another part of the approach is to remove the constraints which are normally associated with systems development (in production) where time schedule and budget considerations usually militate against a systematic test and development program. A number of critical design problems have been set forth for investigation during the entire lifetime of the Advanced Development Program. The first of them uses a bomber aircraft wing carry-through structure as a baseline.

Now for the philosophy on this ADP. We had to set forth some guidelines as to what we want done and so we picked eight technical areas which have to be carried all the way through the whole program:

> Fracture mechanics will be emphasized at all times across the board;

> Structural materials shall be exploited and evaluated with special consideration for heat treatment and improved processes using titanium, aluminum, and steel as basic candidate materials but not excluding other metals or certain mixtures of metals and composites;

> Design criteria will be somewhat more severe than in the baseline structure;

Structural design concepts shall employ safe life and damage tolerance considerations applied to primary structure which is vital to vehicle integrity;

Structural analysis methods shall incorporate at least finite element and numerical solution methods for analysis and optimization of structural configurations, including fracture and damage-tolerant analysis;

Manufacturing methods shall include any new basic and secondary methods which are believed to meet the objectives of this program;

Non-destructive inspection shall be evaluated and exploited, insuring inspectability within predetermined confidence limits;

Information transfer, last but not least, is particularly emphasized at all times for the rapid education of the technical community and not just the contractor.

These eight technical areas have to be considered throughout the four phases of the program: preliminary design, detail design, manufacturing, and full-scale test and evaluation. The idea of the program is to be as diverse as possible in the approach to the problem. The requirements call for starting out in the preliminary design phase with six concepts which will be weeded down until finally one is selected for manufacturing. Originally we envisioned that this cycle would be done once and, if it were so indicated, it would be done over again as an iteration. It remains to be seen whether iteration can be carried out or whether the costs will be too high.

In conclusion, I would like to describe why we feel that the ADP is the way to go rather than to continue such effort through normal systems development. If you have a problem in a system, you are limited in your materials selection, your criteria are fixed for you, you have to use proven design concepts and conventional manufacturing methods, inspectability is a secondary objective, and testing and reporting are kept to a minimum due to time and money considerations.

Fabricability, inspectability, reliability, all these -ilities represent the things about which the presentations by Dietz and Shuler were concerned. In the ADP, we hope to improve all these -ilities up to the next higher level. We will have many more choices for materials. We can use innovative criteria. We can use the best possible design concepts. Inspectability will be a prime objective. Fabrication can consist of variable types, whatever we happen to need. Testing and reporting will be optmized.

With this type of a systematic approach, we should be able to advance the state of the art in structural design and the utilization of structural materials.

SUMMARIZING REMARKS

CDR F. L. Cundari
Naval Air Systems Command

It is not easy to summarize these two days of conference. I was particularly impressed by the spirit of free discussion of the many technical problems existing in Aircraft Design. Most importantly, I have learned that there is a considerable amount of work to do in this area.

In this short time, two days, we did not resolve the fundamental design problems. We did not have the time to study in depth all of the basic issues. However, we did expose most of the problems. It is important that we capitalize on all the issues which we did discuss. In order to provide a reasonable avenue for the conclusive discussion of the salient points, I am recommending that a short follow-up questionnaire be mailed to each participant. Hopefully you will respond with at least the same clarity with which you have been able to express your points so far. Each participant would have to address in more depth the principal issues which have concerned us here.

What are the basic issues? What are the fundamental problems? How can we correct them?

The methods for basic static structural design are common place. The procedures have been utilized for years. However, we have learned that these techniques are not sufficient. Structural design procedures must be improved by incorporating fracture and fatigue analyses tools into preliminary design.

A second major area highlighted at this session is the contractual procedures of the United States Government. These rules or regulations cannot be changed overnight and are not a direct issue of our conference. We must, however, realize the impact of the contract regulations on the design technology. We are not going to change government contracting overnight nor are we going to change the industrial response to these procedures any faster. We will always find it difficult to conduct the requisite engineering design, and evaluation of aerospace vehicles when the government underestimates the task or top management underbids a job.

Let me quickly summarize those points which have been discussed that have an important impact on the technical details of structural design.

1. Fatigue and Fracture

There is a lack of knowledge concerning the basic physical
fundamentals of fatigue and fracture. It appears that
the aerospace industry is utilizing some engineering
approaches to the application of these disciplines
in design. Most likely these procedures are post
design reviews and often introduce costly modifications
or redesign rather than original criteria. So the
fundamental physical knowledge must be obtained and then
it must be converted into sound design principles.

2. Stress Corrosion

It is a difficult task to introduce the current basic
knowledge in this field into a methodological design
procedure. Presently it seems to be either a marvelous
skill employed by aerospace designers which sometimes
approaches a level with pure mystic overtones - black
magic. Often this important technical detail of possible
stress corrosion is not considered until a failure
occurs. This aspect of structural/material failure
must be considered when selecting materials and
configurations in preliminary design as well as the
whole iterative design procedure.

3. Hydrogen Embrittlement & Fretting Fatigue

Closely coupled with the first two problems are the
aspects of hydrogen embrittlement and fretting fatigue.
Both areas have not been fully characterized nor
quantified to allow the development of design procedures.

4. Structural Safety Factor

Reduction of the Safety Factor has been proposed in a
few discussions. It would be extremely difficult to
lower the factor of safety currently used in the
military. Only if a sufficient population of data
concerning the component or part in question is
available, will the service organizations be able to
accept a lower safety factor.

5. Material Characteristics

A more comprehensive data bank of material charactersitics
must be provided. This data must be in a form which is
easily interpreted and used by the design engineers. This
information must include variations with temperature,
both equilibrium and high gradient states; the effects of

the environment including short and long term exposures; and of course, fracture and fatigue characteristics, as well as the fundamental modules and strength data.

This aspect of material characteristics is even more important for the new composite materials, which may have many different reinforcing fiber orientations, matrix materials, fiber types, and lamina thickness.

6. Materials Processing Control

The control of the material manufacturing process has become an extremely important aspect of air vehicle construction. Manufacturing techniques and quality control capabilities must become a part of the basic design philosophy. The heat treatment and aging processes must be fully characterized to reveal the resulting effect on the material structural properties and the effects on the fabrication quality and cost.

7. Service/Operational Data

There is an important need to obtain more sophisticated and accurate operational usage data. This needs to be coupled with "lifetime remaining" type testing of critical vehicle components. Sufficient engineering knowledge must be available to develop the cumulative damage analysis tools for aircraft and missile design.

It may be possible that the new emphasis being given to vehicle prototypes will allow new, untried structural innovations to be adequately demonstrated. This approach could permit a better evaluation of alternative candidate design of some major components.

8. Cost Benefit Analysis

The idea of incorporating the elements of operations analysis or risk analysis into the initial vehicle candidate designs is a step in the right direction. However, the objectives of a national defense force are not usually oriented towards pure dollar loss risks or continual profit margin minimums. The goals are oriented towards weapons performance superiority on a competitive basis. Therefore, we should be developing performance benefit analytical tools that evaluate the potential military combat benefit available for the cost of the candidate designs. This is more of a cost-benefit rather than a loss-risk type of analysis. It appears as though we should attempt to apply some of the techniques that have been developed in other fields and incorporate them in the design phases of the basic airframe.

There are many other important considerations in the design of air vehicles. I have attempted to list only the major items that have been discussed at these sessions.

The objective of this conference was: "to recognize those fundamental design problems which all aerospace companies have in common, to clarify these aspects, and to indicate priorities and guidelines for a methodological development". I feel confident that we have discussed the problem areas. However, I believe we have yet to determine all of the approaches for the development of a methodological design process. I hope that our discussions will be continued until this comprehensive airframe design process is established.

I wish to extend, again, the thanks of myself and everyone associated with the Navy and the Postgraduate School to all participants. Your expert knowledge and your dedication for improving our engineering methods have been exemplary. I know that you will all be willing to participate in the development of this design methodology.

I particularly want to extend my warmest appreciation to Professor Ulrich Haupt for organizing and conducting this conference. His extraordinary interest in airframe structural design has kept the direction of the meeting pointed towards the important goals of this conference and he has been a guiding influence to all of us.

Thank you.

SECTION II

SOME UNSOLVED QUESTIONS AND VARIOUS COMMENTS

SECTION II

SOME UNSOLVED QUESTIONS AND VARIOUS COMMENTS

Discussions during the symposium began along the lines of
the designer's problems as indicated by W. C. Dietz and shown in
Figures C-39 and C-40. Some of the comments were made during
the symposium and some more thorough comments continued coming
in afterwards. A few of them came from engineers in industry
who did not participate in the symposium but were interested
in the subjects.

The questions which were posed are of the kind for which
there is no ready-made answer. The comments -- no matter how
contradictory they sometimes are -- may help to stimulate thinking
about some of those problems which can be anticipated but are not
yet clearly visible. After questions have been formulated and a
discussion has been started, it becomes easier to clarify one's
own thought. Questions and comments in this section should be
considered from such a perspective as somewhat of an opening bid.
No claim is made that they represent a complete and thorough
probe.

1a. Is a concerted effort made to establish a quantitative plateau of acceptability for risks as a guideline for the engineer?

Very limited. Some in establishing gust criteria, some in repeated load and allowable stress areas.

* * *

An effort is made but in qualitative terms rather than quantitative. I, among others, have always felt that each structural component should be assigned a level of risk based upon the consequences of failure and/or the probability of experiencing the critical conditions. Thus, we would have a variable margin of safety throughout the airplane. This has never been done to my knowledge -- though it has been discussed.

* * *

No. Existing specifications talk about fatigue life, fail-safe and safe-life, and structural integrity but not risk. Risk is a bad word that implies failure. Airlines specify a fatigue life for design of primary structure sufficient to "guarantee" a "crack-free" structure. The contract specifies how many flight hours and/or number of landings are required. If the structure "cracks" before these values the manufacturer must repair it at his own expense. The FAR regulation requires a fail-safe strength of 80 percent limit load as the ultimate fail-safe strength. This implies that the residual strength can drop from a positive margin of safety of greater than 1.5 limit load initially to a minimum of 80 percent limit load at the end of a structural maintenance interval and is still acceptable. Of course the manufacturer tries to provide an inspection interval small enough to discover cracks before the strength drops to the 80 percent limit level. Also, the manufacturer tries to provide crack stoppers and a low enough 1 g stress level so that the cracks grow at a slow rate rather than a fast crack growth rate. All these precautions help to reduce the failure probability. However, any loss in residual strength increases the possibility of an occasional load of greater than 80 percent limit load to 1.5 limit load exceeding the residual strength. No failure probability is specified although, as was shown in Mr. Fischler's slides, a loss in residual strength means an increase in the probability of catastropic failure.

Although some attempts have been made to specify the probability
of failure, (i.e., Air Force "Rational Probability of Failure
for the Fleet" and Lundberg's 10^{-10} Structural failure rate
per hour), a "quantitative plateau of acceptability" must
be established which allows "visibility" without excessive
structural weight increases. "Visibility" can be achieved
by comparing the failure probability of successful past
designs to advanced future designs and trying to include
the important variables in such a way to yield a lower risk
with new materials and types of construction.

*   *   *

No. It appears that more static and fatigue testing of
elements, components and entire vehicles coupled with a
better assessment of service life failures will be required
before we can define "acceptably low risk" quantitatively
for the designer.

*   *   *

No -- a risk must be related to factors over which the engineer
has some feel for and control, such as structural weight,
cost, life, etc. Methods of risk analysis haven't been
developed to the point where this can be done in the time
span that most technical decisions must be made.

*   *   *

1b.   What distinction should be made between commercial and
military fields?

A very considerable distinction should be made if one can
accept the idea objectively that a big difference exists
between the consequences of failure on a 250 passenger
airliner and the consequences on a one or two man military
aircraft equipped with in-flight escape devices. This
means that design for military aircraft can and should be
based on probabilistic considerations of loads and allowables
which accepts the possibility of x failures in y hours of
operation.

*   *   *

Commercial vehicles should have a lower failure probability
goal than now exists. The air transport of civilians risk
should be comparable and less than other means of transport
(i.e., car, bus, train or boat). Also, with the increase
in the use of aircraft, (growing at an average rate of 15
percent per year), failures must be fewer to physically
maintain the same quantity of failures per year. In
addition, with the advent of jumbo jets that carry twice

the amount of passengers as the DC8's and 707's, a failure
which results in a total passenger loss, will be so large
a disaster that it is unacceptable even if less than 50
percent as frequent. A purely economic consideration is
the large financial loss from the law suits that result
and the high cost loss per vehicle.

The high failure rate of recent military vehicles has
resulted in the loss of competent military personnel that
should not be measured in dollar value. The required
additional development costs and repairs to correct their
deficiencies after they occur has been unacceptable.
Although less publicity is given to military failures, the
loss has reduced our military strength to an unacceptable
level.

Therefore, the failure risk of both commercial and military
vehicles must be reduced substantially.

Since commercial vehicles have a desired life of approximately
20 years with a higher yearly utilization than military
vehicles with a desired life of approximately only ten
years, then commercial vehicles must have a lower failure
probability to account for these factors. Also, if the
loss of a human being, whether a civilian or military, is
at the same level, then the risk exposure per passenger
mile should be the same. Since military vehicles normally
carry a much smaller human payload than commercial, over
an average shorter range, (since fighters are short range),
these factors should also be considered. Military vehicles
therefore could be designed for a higher risk of failure
structurally if the acceptable criteria is to match the
failures structurally per commercial passenger mile.

Military vehicles, in general, are designed for higher
load factors than commercial vehicles (i.e. a fighter is
designed for 7.33 limit load factor and a commercial
transport for approximately 2.5g). The load factor occurrences
per mile of a fighter however are higher than the commercial
vehicle. A higher design load factor for a fighter results
in a lower 1 g stress level. Therefore, the rate of crack
growth per hour of flight for a fighter is usually lower
than for a commercial transport. Therefore, the loss in
residual strength for a fighter, in spite of the higher
frequency of loads, is less than for a commercial transport
per hour of flight. These factors also must be considered
when comparing the expected failure rate. To overcome the
disadvantage of such a high design load factor for a
fighter, high strength steels or titanium are frequently
used. A slight reduction in strength allowables, (from
260,000 p.s.i. to 200,000 p.s.i.) for steel would increase

the fighter life substantially. In conclusion, design of
the transport or the military vehicle is very dependent
on the desired life and the environment expected. A
proper design, considering fracture mechanics could result
in an efficient design for either aircraft with an equivalent
probability of failure (same failure rate per passenger
mile).

*   *   *

The goal for both military and commercial aircraft should
be to obtain a low level of risk. However, the trade-offs
for military and commercial aircraft are usually different.
For commercial aircraft risk must be weighed against cost,
whereas for military aircraft risk is weighed against
performance primarily. Therefore, more costly solutions
to design problems become necessary for a military
aircraft as compared to a commercial aircraft.

*   *   *

... Military should be willing to accept risks to help
assure early availability of high performance systems.

*   *   *

1c. What is the trade-off between increased probability of
failure and increased performance for military aircraft?

None. Higher probability of failure should not be accepted
to increase performance except in aircraft deliberately
designed for short life or to meet national emergencies.

*   *   *

... Many failures are a function of the poor selection of
materials for structures, controls, propulsion, and poor
inspection procedures. All these areas' reliability must
be improved plus the use of new materials that are more
damage tolerant (graphite epoxy at low stress levels),
before the risk of designing for higher stress levels to
reduce the structural weight and increase the probability
of structural failure to get increased performance can be
considered. A systematic failure probability analysis
which has visibility and is accepted as a standard for
comparison to get the failure rate lower must first be
accepted and tested by actual flight performance before
considering the trade off of parameters.

*   *   *

Obviously, increasing the probability of failure should
result in a lighter, less expensive aircraft, but as the
probability of failure increases, the overall cost of owner-
ship of x aircraft in service will increase.

*   *   *

Quantification not really possible. The military risks
unknowns of new environments and new materials. Best
thing would be to provide for tolerance of difficulties
regardless of source as opposed to acceptance of unknown
risks.

*   *   *

1d. Should different levels for probability of failure be
specified for different types of failure -- partly for
psychological reasons?

Yes, but first define "failure consequences." The probabilities
should be tied not so much to the type of failure as to the
effects of the failure.

*   *   *

Yes. Structural failures are unacceptable by the public.
The Martin 202 and Comet were types of structural failures
that the public would not accept. Therefore, structural
failures should have a very low value so that their
occurrence should occur once every 20 years, or based on
the expected passenger miles projected for the 1980's,
values of structural failure that result in the loss of
only one jumbo jet aircraft from a structural failure in
20 years.

Failures occur most frequently from pilot error, power
failure, system failure, gear failures, and structural
failures. To improve the overall failure rate a study of
these failures must be done to uncover why the failures
occur. Setting different levels of failure is too premature.
First a better data collection and failure analysis must
be made to discover the underlying reasons for failures
that need correction.

*   *   *

Yes, depending on mode of failure, e.g. partial failure
in fail-safe structure or slow crack growth in safe-life
structures detectable by routine inspection and repairable
will permit higher stress levels and conceivably better
systems performance.

*   *   *

The probability of failure can not be predicted with any
degree of accuracy at the present time, so it is not
feasible to distinguish between different types of failure
at the present time. Certain parts which can cause loss
of an aircraft should be considered differently than parts
that can be lost without seriously endangering the
completion of a flight or mission.

<div align="center">*   *   *</div>

The consequences of a local buckling failure and a tensile
failure can be vastly different with respect to safety
in flight and therefore we might tolerate a higher
probability of buckling.

<div align="center">*   *   *</div>

Clever design should be able to provide a forgiving or
tolerant enough structure to make this question not
relevant.

<div align="center">*   *   *</div>

1e. How can the climate in the public and in congress be changed
from an "anti-technology fad" to an understanding of the
engineering process and the risks connected with new developments?

You answered the question. If it is a "fad", time alone
will improve the climate.

<div align="center">*   *   *</div>

The current "anti-technology" fad, particularly in Congress,
is in part related to an inability to communicate in a
meaningful way. Tangled up in the problem are the economic,
environmental, and social problems that have led to a
near breakdown in meaningful exchange. It sometimes
appears that in major weapons systems developments the idea
is never clearly brought out that there are technical
problems in such developments that challenge the then current
state of engineering development and understanding, and
that their solutions cannot be anticipated at the time a
contract price is negotiated. Both the developing
organization and the purchaser (generally, a DoD agency)
will have to be more candid in discussing (1) possible
problem areas, (2) the stretching of the state of the art
to solve the problems, and (3) the risk of success or
failure. Congress will have to learn how to listen, how
to question, and how to evaluate with a minimum of political
haymaking.

<div align="center">*   *   *</div>

By establishing a reputation for telling the truth. Better
communications with the legislators at all levels. How
about a monthly newsletter from each Contractor reporting
major   test, flight, or service failures. With an
explanation of their cause, correction and technical
implications -- candid to the point of admitting errors
of commission and omission. In this some of the calculated
risks in design could be explained.

<center>*   *   *</center>

Better presentation of the facts and alternatives by news
media and television reporting are needed. The "anti-
technology" factions seem to get their points well
publicized even when there is little technical basis for
some of the points. The publicity during the SST debate
is a case in point. The public and many Congressmen seem
to have difficulty distinguishing the difference between
someone's theory and results based on experimental evidence.

<center>*   *   *</center>

By the military services, the National Science Foundation,
the President's Scientific Advisory Board, the Ford Foundation,
the AIA, the AIAA, the SAE, ASNE, ASTM and other groups
being contacted to have joint meetings to decide on a course
of action. Probably a T.V. public relations concern would
have to help, plus newspapers, plus the college boards,
and large corporations that need the technology.

<center>*   *   *</center>

I don't believe patience is enough. We have to use the
media to play up the good side. AIA, ATA, AIAA, SAE,
etc. all are missing the boat here.

<center>*   *   *</center>

Hostility and fads can only be neutralized by education...

<center>*   *   *</center>

1f. How much does the new outlook toward fatigue, fracture
mechanics, and service life affect our traditional attitude
about limit and ultimate load factors?

The detailed consideration of fatigue, fracture, and service
life in the design of complex aircraft systems can lead
to better and safer structures. It may reverse the trend
toward continued utilization of ultra-high strength materials
of dubious flaw tolerance, which is the direction that
ultimate and limit load concepts leads. It also is bound
to stimulate the detailed materials and processes research
necessary to develop flaw tolerant materials.

<center>*   *   *</center>

These are new tools which indicate that a critical crack
length must never occur, that the decrease in residual
strength due to a crack increases the risk of a load
exceeding this strength and therefore increases the risk
of failure of the entire vehicle. To decrease the risk
of failure the margin of safety of 1.5 should be studied
further to include the time factor of rate of crack growth,
loss in residual strength, and probability of failure.
Notice that for titanium and graphite epoxy an ultimate
strength to one "g" stress must have a factor of more than
3.75 (1.5 x 2.5 g) for a transport critical for the limit
load factor for maneuver of 2.5.

<p align="center">*   *   *</p>

This is a healthy attitude provided it doesn't go overboard.
Structural engineers have always considered fatigue,
fracture mechanics, etc., at least since the 1930's. But
it takes a political hot potato like the F-111 and its
problems to dramatize the problem. We will always have
to be concerned about static loads -- but with more and
more emphasis on probability of occurrence.

<p align="center">*   *   *</p>

I don't think that fatigue, fracture mechanics, and service
life is changing our attitude about limit and ultimate
load factors at the present time. The limit and ultimate
load factors used provide a sort of baseline configuration
from which to work. Structural weight must then be added
to meet other requirements such as structural life, damage
tolerance, etc. It will be some time before structures
can be designed on the basis of failure probability rather
than on the basis of an ultimate load factor of 1.50.

<p align="center">*   *   *</p>

Design for service life makes the ultimate load factor
concept obsolete and of little value. The design must be
based on a load spectrum that includes a limit load
defined as the load the vehicle must sustain with some
prescribed probability of success on the last mission
in its total service life or in each inspection interval.

<p align="center">*   *   *</p>

I'd say anyone that thinks traditional factors should be
attacked does not really know the vagrancies of our
knowledge base. I believe in rationalization but not
exploitation!

<p align="center">*   *   *</p>

It has not "sunk in" that improved methods of analysis,
materials with yield to ultimate strength ratio different
from aluminum alloys, better load prediction methods,

process control etc., could obsolete the old traditional ultimate (ignorance) factor. Substantial improvements in systems performance and safety can be envisioned if arbitrary load factors are replaced with a design criteria system taking critical load or load spectra and critical strength or failure mode into account. All that can be said in defense of tradition is that such system is not yet developed. They should however not block the work needed since tradition is as incompatible with engineering as it is with military science.

*   *   *

2. Regarding development of metallic materials.

2a. How much can reliability be improved by close control of processing?

Substantially. However, a check is needed at the source of the material, before it is fabricated, and after it is fabricated, to assure compliance with the specified process.

*   *   *

A quantified answer to this question is difficult to come by. However, close control of processing, if it is economically acceptable (which is part of the risk analysis problem), can be expected to improve reliability, and may result in higher and safer design stress levels. This is really the reason why most major aerospace companies maintain highly motivated and competent materials and processing personnel.

*   *   *

Reliability can always be improved by closer control, but getting the producer and processer to improve control, particularly with no real definition on how close is close, is impossible. Therefore, the only practical way of improving reliability in metallic materials behavior is to develop more definitive NDI/NDT techniques along with fracture mechanics analysis to predict the influence on structural strength.

A practical way of establishing meaningful producer/ processer controls would be to define a financial responsibility. Presently, if a producer/processer gives you "bad" material, his only responsibility is to replace the material, not the configured part that usually costs an order of magnitude greater than the raw material. If

the producer/processer can be made to bear some of this
expense, control will improve dramatically.

* * *

Modest improvements could be expected in the order of 10%
less improperly processed parts. More destructive testing
of actual production parts is far more effective than
200 or 300% visual and non-destructive tests or the
examination of standard test coupons. Particularly,
applies to welded, bonded, cast, and forged parts. Is
expensive in practice but compared to the loss of a
vehicle, extensive investigation, or proof tests relatively
cheap.

* * *

Perhaps a great deal for structural applications which
are not tolerant of flaws. Process controls have to be
compatible with fundamental reliability requirements and
generally are so restrictive as to be premissive of no
real variation.

* * *

Substantially.

* * *

2b. Has there been a wrong emphasis in the development of
metallic materials on increasing strength-density ratios
instead of improving fatigue and fracture toughness?

It is not entirely a case of wrong emphasis. There has been
a real trend in the development and use of high strength-
density metallic materials, in accordance with established
design practices, in order to solve, presumably, the
mutual problems of performance and weight. While fatigue
and principally fracture become more critical with this
trend, it is only recently (5 years or less) that acceptable
organized design procedures have evolved, particularly to
handle the fracture and crack propagation problems. Because
of the attention that has been paid in characterizing the
phenomena and in developing rational design procedures,
less attention has been given to the alloy development
activity. Inclusion of a damage tolerant design requirement
in the B-1 is certainly going to stimulate this kind of
research, in view of the gains already evident.

The slowness in the improvement of materials for fatigue and
fracture resistance probably is more closely tied to
availability of funds than to lack of ideas. Alloy

development for structure sensitive properties, such as fatigue and fracture, is a time-consuming and costly operation. In comparison with construction materials and automotive materials, base sales of aerospace materials are small enough at present to discourage large-scale research of this kind without massive interest of DoD and NASA agencies in the form of funding.

<div align="center">*   *   *</div>

The fatigue and fracture mechanics tools are relatively new and they certainly need to be exploited to the fullest. However, new metallics should be judged considering many factors. The strength-density factor is only one ingredient and should also be obtained. Frequently the fatigue strength after 10^8 cycles is a percentage of the F_{tu} value. Also, the K_{ic} value is often a percentage of the F_{tu} value. The residual strength, (initially F_{tu}), decreases with crack size, stress range, stress intensity factor (K_c), and type of construction.

The slowness of improvement in fatigue, or the recognition of a fatigue problem, is caused by the slow development of the finite element approach to discover the state of stress causing a fatigue problem and/or a fracture mechanics problem. As soon as the stress field is known the fatigue problem becomes obvious.

<div align="center">*   *   *</div>

Probably, although I think it is pretty much recognized today that tensile strength is an inadequate description of the usefulness of a material. It does provide a "floor" for comparison.

<div align="center">*   *   *</div>

Possibly -- we are approaching minimum gage restraints in many applications. Therefore, higher strength, per se, may not always be needed. Most of our systems are fatigue and crack growth critical now. Since fatigue, crack growth and fracture toughness do not generally increase with increasing strength, more emphasis should be placed on these properties in developing new and improved materials.

<div align="center">*   *   *</div>

This emphasis was probably proper in the past but in the future we must seek a proper balance between strength, fracture toughness and crack growth resistance of materials. This balance depends on the particular application. For parts with long cyclic life, the crack growth resistance is the most important characteristic (and the least studied and understood). The high strength of a material may not be usable in such a case because the material has a low resistance to crack growth or low fracture toughness.

<div align="center">*   *   *</div>

If this is a true assertion, my answer would be yes. However, designing to accommodate any particular deficiency is an alternate if attributes more than balance deficiencies.

*   *   *

Yes.

*   *   *

Yes, I think so, since most of structural problems are associated with repeated tension loadings and little advantage can be taken of the high strength properties. Higher fatigue properties would give us a greater advantage.

*   *   *

Yes. Emphasis on strength/density ratios has been beneficial in developing high strength materials up to the point where toughness and fatigue transitional effects become dominant. The strength/density emphasis was also necessary in order to highlight the need for materials where fracture toughness and fatigue keep pace with strength. However, strength levels have presently reached the point where emphasis must be placed on fracture toughness/fatigue criteria in order that these materials remain practical for utilization with a degree of confidence.

*   *   *

2c. How much could be accomplished with additional funding for the development of metals? What kind of incentive is required?

To take an example in regard to what can be accomplished with additional funding for research in metals, I cite recent B-1 experience with Ti-6Al-4V alloy. This alloy, the workhorse of the titanium family, has a minimum fracture toughness variously estimated to be in the range of $30 - 40$ ksi-inch$^{1/2}$. With the imposition of a fracture criterion for the airplane, funds from the Air Force, and good work, North American Rockwell now is specifying the same alloy with a minimum fracture toughness of 70 or 75 ksi-inch$^{1/2}$. The improvement in crack tolerance, by doubling the fracture toughness, is vitally important to the airplane. The ingredients or incentive to accomplishing this were (1) the requirement of fracture-safe design, and (2) funds to accomplish the development work.

I would expect there are still improvements to be gained in the upgrading of fatigue and fracture behavior in metallic systems of aluminum, steel, and titanium. There probably will be modest percentage improvements for the first two alloy systems. I prefer to think that titanium has yet a way to go in useful alloy development.

*   *   *

The diffusion bonding of metallic parts into a good compression and fatigue resistant structure with light weight foils has been attempted but the development costs are high. If the incentive of a bonus for higher strength small components is made with all the development costs sustained by the government, then metallurgists and structural engineers would devote their effort in that direction. Heat testing a matrix which is diffusion bonded and later etched out to reduce the wall sizes also needs to be tried with small components.

Adding small quantities of high strength materials to low strength materials (Beryllium, Boron, or graphite in aluminum or magnesium or glass or an epoxy matrix).

Making sandwich structure with high fatigue strength of the core with high strength edge attachments could be developed.

Etching of covers and diffusion bonding to metallic cores to make up light weight sandwich structure has been partially successful. However, a development program to set up a production method using Vee cores or double Vee cores or light weight channel sections has not been successfully tried. The etching can provide fail safe crack stoppers and end attachments.

<div align="center">*   *   *</div>

The sky is the limit. Physicists say we are realizing maybe 10% of atomic load capabilities. Profits are our incentive -- government should give us the opportunity.

<div align="center">*   *   *</div>

Magnitudes are not the most important -- put the money where the capability exists and the incentive exists, i.e., where the development falls into the company's normal sphere of business.

Theoretical considerations along with practical attainments, say that steels with 500 ksi, aluminum with 150 ksi and 300 ksi titanium are totally feasible.

<div align="center">*   *   *</div>

Funding should be concentrated in the critical, now better understood areas of fracture strength, fatigue strength, creep strength, and in stress corrosion and materials process control.

The probabilities to obtain significant improvements are surprisingly good, judging by recent breakthroughs like the improvement of creep strength of titanium alloys at high temperatures, etc.

<div align="center">*   *   *</div>

I am a composites enthusiast but I see aluminum and steel
continuing to be the workhorse structural materials for the
next twenty years even if no significant improvements in
properties are made. As for titanium it will continue to
be used where the operating environment requires it. I
cannot assess the probability of improvements.

*   *   *

2d. How much can we learn from the fact that corrosion resistance
and crack tolerance of aluminum alloys were improved considerably
in the last few years by a relatively small effort after a
stagnant situation for more than a decade?

I am not sure of the phrase "relatively small effort".
All the DoD agencies and NASA, as well as the aluminum
companies, have spent considerable sums of money in
developing alloys and tempers of high-corrosion resistance
and crack tolerance. However, in answer to the questions,
it was in part a response by a knowledgeable industry to
a known set of technical deficiencies. It represented also
an approach by the industry to protect its markets in the
face of newly emerging structural materials, such as
titanium alloys and composite materials. Finally, it
probably was stimulated by some urgent prodding by DoD
and NASA personnel, as well as the aerospace community.

*   *   *

The theory for crack propagation was available and the
finite element method of analysis was developed in the last
decade to confirm the theory and investigate the problem
further. Also, most important, the desire to find out
how to practically deal with the problem was present.
Many large size panel and fuselage components were tested
at considerable expense at Douglas for the DC8, DC9, and
DC10 to develop crack tolerant structure. Stress corrosion
has also been studied and corrosion resistant coating have
been successfully developed when the problem was recognized
and needed solving.

*   *   *

This suggests to me the classic situation that prevails in
many disciplines. A period of little apparent progress
during which research in related areas continues. And
then a period of rapid growth made possible by the synthesis
of several ideas that have incubated and reached maturity.

*   *   *

During that period, objectives were not clear, research was done by all because it was fashionable and the Government paid for it -- achievement was not the goal used to measure success, only budget and schedule.

*   *   *

State the real problem and ingenuity can find a solution. In this case with minimum subsidy.

*   *   *

We may learn that progress is not so much dependent on magnitude of funding but on clear definition of problems which must be overcome and on time to solve them.

*   *   *

2e. In view of the small market in aerospace compared to the vast automotive market, is there a possibility to apply aerospace R & D toward reducing automotive manufacturing costs and to establish a broader base for development funding?

We are discussing our methods with a leading automotive company and if we can "sell" them on our advanced methods we expect them to do more R & D in the use of applications common to both and thus broaden the base for development funding eventually. The first step, and only step we are now pursuing, is to teach them advanced methods of structural analysis.

Remember that initial tooling and manufacturing costs are the big items, a pound of weight saved is not as important to an automotive concern. We must convince them that the pound saved will yield fuel economy, less braking, smaller brakes, and the ability to put the pound into items that will increase the life or comfort level. How much they are willing to pay for this pound is not known. It is not uncommon for a long life advanced aircraft to use a $300/pound breakeven value. Can an automotive firm pay that price? A 1500 pound automobile, if all items were assessed at $300/pound would cost $450,000; a 1500 pound auto is more like $1.5 per pound. Therefore, it is only in the analysis, manufacturing, and tooling development we can help the automotive concern if they are willing to use materials we are familiar with -- aluminum, titanium, super alloy steels, and possibly composites.

*   *   *

Should the question be turned around? Is there a possibility
that automotive R & D with its broader base of development
funding can be slanted to give benefit to aerospace? The
area that is common to both is in the manufacturing
process field; forming, metal cutting, joining, shaping, etc.
I assume that manufacturing people in aerospace stay pretty
close to developments in the automotive field.

*   *   *

Unlikely, since our objectives do not coincide. However,
we are each being forced to consider the others point of
view, namely, reliability, light weight, and safety of the
aircraft industry and the low cost, high automation, and
prototyping of the automotive industry. This is an exchange,
even now, of each others R & D developments, ECM, EB welding,
reenforced plastics, turbine engines, numerical control,
molded plastics, fasteners, adhesive bonding, etc. It
does not appear that the automotive community would want
to underwrite R & D in aerospace since it would not be
oriented directly to their needs.

*   *   *

Not probable because aerospace objectives are totally
different than automotive, i.e., aerospace -- high
reliability, long life, low production, hostile environment;
automotive -- mass production, planned obsolesence, etc.

*   *   *

The economics of the two industries is so different that
this possibility seems very remote in a useful way to me.

*   *   *

Sure!!! But who is doing it? I think the government should
provide leadership. Our professional societies have
struck out so many times.

*   *   *

I do not think so. Requirements are generally too different.

*   *   *

3. Regarding present trends in fatigue criteria

3a. Have we challenged the designer to recognize problems and
limitations applicable to fatigue, particularly with regard
to processing and inspection capabilities?

Many good designers are aware of some of the important
factors. However, often these designers are not as close
to the detail design as they should be. Therefore, for the
average designer a handbook should be prepared to give
him the necessary process and inspection specifications
to <u>note</u> on the drawing.

*   *   *

I think the designer is unaware, generally, of the potential
variation that processing has on fatigue-crack propagation,
though somewhat better informed on fatigue behavior.

*   *   *

I think our designers are fairly mature with respect to
recognizing the differences between paper designs using
paper properties and the real world. This is not to say
that a continuing effort is not required to maintain and
enhance this awareness -- especially for the relatively
new and inexperienced engineer.

*   *   *

Designers are aware of most of the tricks used to improve
fatigue in structure, but they can only call out an existing
specification to cover processing and inspection -- this
is an area that needs much work.

*   *   *

Designers have been challenged but it went over their
heads. Much more needs to be done in this area. Most
know fatigue problems are detail design and/or manufacturing
deficiencies.

*   *   *

Not yet adequately.

*   *   *

Designers will begin to recognize that flaws are
characteristically present in structures and that precautions
in material selection, processing and inspection are of
paramount importance, particularly where high strength
structures are concerned. Fatigue cracking rates and K_{Ic}
data are presently still research items, however, it is
anticipated that this will reach the designer as a practical
tool in the near future.

*   *   *

3b. How much can be gained from materials application compared
to materials research?

Material development needs both. Separating the materials
research from the materials application is often done for
convenience but the complete package is necessary to
prevent the designer from going from the research results
without the confirmation of an application result.

*   *   *

Materials application of improved state of the art
materials (somewhat nonconventional) will point up the
need for more research as materials and attendant
manufacturing difficulties are encountered.

*   *   *

Both areas are complementary -- one without the other can
only be a repetitive function, not an advancement.

*   *   *

Research results will not be used unless there is an
adequate materials application program.

*   *   *

New applications should be emphasized but not at expense
of research. We must exploit what we have at hand and
at the same time provide for the longer range future.

*   *   *

Wrong question. They should complement each other.

*   *   *

3c. Would tightening of specifications and processing methods
for titanium, for instance, result in consistently finer grain
size and better fatigue properties?

I am not completely certain that we can at present specify
process controls for titanium and other alloy systems
that will optimize fatigue properties. This is one of
the research areas, particularly for titanium, that needs
definite study.

*   *   *

Probably not -- only if finer grain size caused fatigue
improvement and there is contradicting evidence here -- what
is needed is to define the parameters in the metal that
control fatigue behavior before we can "shot gun" an
improvement with any chance of success, not just increase
cost.

*   *   *

Yes. A tightening of specification and processing methods
is desirable. However, the effect on cost should be given
also so that the designer is given a choice depending on
the material application.

<center>*   *   *</center>

Hopefully. I would prefer some better understanding of
what causes what and then refining.

<center>*   *   *</center>

This is needed once processes have been defined and cost
effectiveness is established.

<center>*   *   *</center>

Not necessarily. Might tend to price material out of
realm of practical utilization. Applied research and
development in addition to more stringent specification
requirements are needed. Changes in titanium specifications
brought about by alpha segregation problems have led to higher
prices. This type of defect has only contributed to
rotating engine part failures, so unnecessary tightening
can lead to higher prices and therefore decreased usage.

<center>*   *   *</center>

3d. Realizing that fatigue analysis is not yet an exact
science and that fatigue design has to be based on previous
experience, how can this experience be evaluated systematically?

We have a fatigue checkoff list which is needed before signing
off a drawing. Also, we have put together fatigue problems
in a fatigue course. Unfortunately, the fatigue book
has not always been kept up to date with examples of
failures from other manufacturers. Perhaps the military
services should fund a project for collecting these fatigue
examples which all manufacturers would contribute to
periodically and the services would redistribute periodically.

<center>*   *   *</center>

I assume you are asking how can we learn more from the sum
total of fatigue failures that have been experienced. We
all agree (I think) that we can learn more from our failures
than our successes in material application engineering.
But there is an understandable reluctance to publish the
failures. The most interesting technical papers I have
read have been those that were completely candid about the
wrong turns, the mistakes, and the failures, prior to the
successful solution. Let's encourage more of this through
our professional societies.

<center>*   *   *</center>

One can write a book on this question. There are some
simple elements to approach this problem, some or all of
which are being pursued to some extent. At the design
stage, one's past experience with a similar airplane
provides a link between design loads and real loads, and
between design response (deflections and stresses) and real
response. This information coupled with careful history
on fatigue initiation and propagation in the previous
airplane, provides some semblence of how the new structure
will behave. The ingredients involve evaluation of what
happens in many numbers of a fleet of a given aircraft
during service in order to characterize realistic stress
spectra for the aircraft. Similarly, one monitors the
fleet for fatigue damage. Based on developmental fatigue
tests of structural components, on fatigue of the basic
materials, and on the fatigue test of the entire airplane,
one can compare, analyze, and evaluate real life behavior
and the basic soundness of previous programs and design
philosophy.

<center>* * *</center>

First the variables affecting fatigue life must be properly
identified. Then failures obtained in service or test
would have to be defined using these same variables. These
failures could then be used as data points from which
relationships could be established between the variables.
The relationships could then be used for predicting future
experience.

<center>* * *</center>

This could be the theme of a book and has been on numerous
occasions. I have no handy solution. Design handbooks of
not-to-do's come closest perhaps or, better, best-to-date
approaches could be useful.

<center>* * *</center>

Probably not beyond what to do and what not to do.

<center>* * *</center>

Previous experience mostly allows one to better evaluate
stress concentrations and better define design load spectra.
An attempt to classify K_t for practical structural con-
figurations might be helpful. Also design load spectra
that have been inadequate for certain systems should be
corrected and applied for similar applications on the
next design. Attention should also be paid to those
systems where the design spectrum or K_t has been assumed
too severe by reducing the requirement in the next design.

<center>* * *</center>

4. Regarding current trends in fracture mechanics and fail-safe design

4a. W. S. Hyler's presentation indicated how much scatter can exist for K_{1c} data, even at uniform F_y. This reflects on uncertainties of materials manufacturing, i.e., "process capabilities". What practical conclusions can be drawn regarding processing and specifications?

Further analysis at Lockheed of D6AC K_{Ic} data indicates that the K_{Ic} value is a function of the quench rate obtained during processing which in turn is a function of the thickness of the part processed. The tensile properties in this case were not affected by the quench rate or thickness. The obvious conclusion is then that K_{Ic} type tests must be specified as part of the process control if the fracture toughness properties are the properties that must be controlled. It is becoming obvious that ultimate strength tests can not be relied on to adequately control material properties such as fatigue, fracture toughness, crack propagation, etc.

<p align="center">*   *   *</p>

After further research, a specification which indicates the quench rate, heat soak time for grain size, and the allowable impurities necessary, plus the inspection and sample testing necessary, should be specified versus the coefficient of variation versus the cost comparison for an A, B, and C type value.

<p align="center">*   *   *</p>

Mr. Goepfert of ALCOA, a number of years ago, indicated on the basis of a large amount of testing and statistical evaluation that a specification does not control a process, rather it is the process that leads to the specification. Therefore, the decision to establish in a specification a minimum K_{Ic} level without sufficient knowledge of how a given process schedule influences the distribution of K_{Ic} values for the product can only lead to erroneous values, usually too high. Specifications can provide target values toward which the producer may move by modifying his process. But if no process can be evolved to acceptably meet this target, then it is the process that controls.

<p align="center">*   *   *</p>

1. Don't give up -- keep working;
2. Don't depend on material quality alone;
3. Recognize possible variation and design around.

<p align="center">*   *   *</p>

Extensive work is required to research processing effects
and to establish processes resulting in consistent properties.
It is obviously feasible to get improved toughness.

* * *

As fracture concepts become more prevalent, K_{Ic} criteria
will become a standard spec. item for high strength
materials. This should lead to processing methods for
optimizing fracture toughness. Vacuum melting along with
fabrication techniques designed to minimize anisotrophy
effect will become requisites for obtaining consistently
high K_{Ic}/F_{ty} properties.

* * *

4b. How can material be characterized clearly for fracture
mechanics with respect to processing as well as thickness?

A significant amount of testing is required in order to
establish process capability for fracture just as for
establishing process capability for F_{tu} and F_{ty}.

* * *

The fracture toughness properties must be characterized
according to the variables that can affect the properties,
e.g., quench rate and thickness for D6AC steel. More work
needs to be done on materials to determine what effect
processing variables have on the material properties, so
that better processing specifications can be written.
Sufficient tests need to be conducted so that MIL-HDBK-5A
type allowables can be established.

* * *

The averave K_{ic} value can be tested to determine its value
versus thickness (using a close range in processing and
inspection for the samples tested). From the sample
size the confidence level can be given for each specific
thickness.

* * *

By process specification and more exploratory work in
mixed mode fracture and ultimately design oriented testing.

* * *

We don't know yet, but it must be through experimental
correlation and improved analytical tools.

* * *

4c. What guidelines can be established for trade-offs between higher strength and lower K_{lc}?

The K_{ic} value influences the rate of crack growth and the residual strength versus crack size. A chart which shows the residual strength, rate of crack growth versus K_{ic} with constant values of one "g" stress, desired life, and range of stress or, (σ_L/μ_L), coefficient of variation of the loading, can show the designer the compromise he must make on the value of K_{ic} and Fy (which influences the initial residual strength).

*   *   *

Material and component tests, if done properly, should yield precise results.

*   *   *

There are many parts of a structure where one need not be concerned with fracture. Compression surfaces are one of these. High strength, moderate to low fracture toughness probably provides no problems in these areas and one has the advantage of high strength. Areas in certain aircraft structure offering multiple load paths may be another place that can dictate tradeoffs between strength and toughness, depending upon considerations of risk and reliability.

*   *   *

I should think that highly redundant structures as well as those employing fail-safe design practices could lean more toward higher strengths at some sacrifice in K_{lc}. But as noted earlier, so much depends on the application and the acceptable risk.

*   *   *

The tie-in here is the allowable flaw size that can be tolerated in the structure. If NDI capabilities are such that the flaw size existing in the structure is very small, then F_{tu} or fatigue properties may design the part. However, as the flaw size existing in the material increases, fracture toughness properties become all important. More work needs to be done to improve NDI techniques and the reliability of these techniques.

*   *   *

For each design situation (load spectrum, initial flaw size, and required service life) the relative importance of ultimate or yield strength, fracture toughness and crack growth resistance can be determined and the extent

to which each might profitably be increased or decreased
established. Unfortunately little such analysis has
been undertaken and none of it was evident at the symposium.

$$* \qquad * \qquad *$$

Question whether there is a real trade-off involved.
Design around!

$$* \qquad * \qquad *$$

A good fracture control plan covering all aspects from
basic material to in-service inspection.

$$* \qquad * \qquad *$$

The impact of reducing the strength of a material to
improve K_{1c} would be less if fatigue life properties were
retained.

$$* \qquad * \qquad *$$

1. Establishment of minimum critical crack sizes and
resulting K_{Ic}/Fty requirements.
2. Evaluation of K_{Ic} versus Fty data to determine how
rapidly K_{Ic} falls with increasing Fty and practicability
of controlling material within required K_{Ic}/Fty range.
3. Correlation of cyclic crack growth and K_{Iscc} properties
with critical crack size and NDT inspection.

$$* \qquad * \qquad *$$

4d. Can any conclusions be drawn with respect to fail-safe
design?

Probably, to me, the most unsettling factor in damage
tolerant design is its use with thick section materials,
where the critical crack size may be borderline with
regard to the probability of detecting a crack. NDI
techniques are reputed to be capable of finding quite
small flaws. However, the probability of finding such
small flaws may be equally small in practical situations.
Consequently, in damage tolerant design, it makes some
sense, to me at least, to base propagation life and
inspection intervals on a crack length eminently capable
of being found most of the time, rather than a length
within the minimum bounds of the inspection device.

$$* \qquad * \qquad *$$

Fail-safe design is a design that according to the civil
specification can sustain 80 percent of limit load at any
time during its life time. To insure this, the inspection
interval must be short enough to repair any cracks before
the residual strength decreases to the 80 percent level
or the critical crack length occurs (fast fracture occurs).
With transport vehicles, the requirements also desire
that a full bay crack can be sustained before failure.
Now, K_{ic} influences the residual strength and the rate of
crack growth. It would be desireable that K_{ic} have a
coefficient of varation small enough so that the value
used for predicting the residual strength and inspection
interval, (before a full bay crack occurs), is reliable.
As shown, in Mr. Fischler's presentation, the decrease
in the coefficient of variation, (σ_S/μ_S), increases the
probability of failure greatly especially for a V/STOL
aircraft with a high load spectra coefficient of variation,
(σ_L/μ_L). Therefore, specifications which insure values
of K_{ic} within a narrow range are desireable.

*   *   *

By fail-safe design, it is implied that cracks, if they
occur in the structure, will be found before they become
catastrophic. Therefore, heavy reliance is placed on
finding these cracks during routine inspections. To
exploit the higher strength potential of metallic materials
will require a substantial improvement in inspection
capabilities to achieve the same reliability as with
current materials at lower strength levels.

*   *   *

Fail-safe design has many merits, but it was not discussed
in any depth at the symposium.

*   *   *

I sure do not agree with Fig. C-40 regarding fail-safe
problems. It reminds me of my dad's philosophy that "you
only get out of things what you put in". If one sets
out to prove that fail-safe designs can be devised that
have problems, he can sure do that. If he intentionally
configures so as to minimize "possible" problems, they
will not arise.

*   *   *

Fail-safe design should be employed wherever feasible,
where not fracture control by use of tough materials,
proof testing, etc., should be employed.

*   *   *

There is more inherent fail-safe capability in a structure
than will ever be calculated. On military aircraft there
is great danger of going overboard with fail-safe require-
ments unnecessarily. This has been shown by aircraft
that have returned with significant battle damage in
components that provide no visible fail-safe load path.
On military aircraft the fail-safe concept should be
utilized so long as it does not increase airframe weight.

*   *   *

5. Regarding evaluation and application of structural metals

5a. A report NMAB-246 was published in 1970 under the title:
An Approach for Systematic Evaluation of Materials for Structural
Application. It contains an outline for a data information
system requiring data banks on material properties, material
evaluation techniques, and applications analysis. Is there
a follow-on area where fruitful concepts could be developed,
e.g.

failure analysis to develop new test techniques;

case history development to support new test needs;

or possible contributions of information centers?

Yes, a follow-on area where failure analysis theories can
be confirmed by tests would be desireable. I would like
to test materials, find out their F_{tu}, F_{ty}, and K_c values
from coupon tests, obtaining enough tests to get a high
confidence level for their coefficient of variation.
Then, using parts of the same sheet as the coupons come
from, make up a test component of structure. I would
test some with and without an initial crack at different
mean stress levels and different stress ranges, periodically
checking the residual strength by failing some of the
specimens. The specimens remaining should be continuously
cycled till every one fails. The actual life, residual
strength, and rate of crack growth should be compared to
the predicted values to determine the accuracy of the
theories. Adjustments to the theories should be suggested
to account for the slowdown in crack growth when the
cracks reach attachments.

*   *   *

I'd like to see an outfit like the Batelle DMIC try out
the ideas contained in NMAB-246 on a selected type of
structure or component. This should not be a big deal but
could serve as a pilot operation.

The Case History approach is good and might be a way of
motivating more reports on failures, as for fatigue.
Perhaps Don Shinn can spark plug a Failure Reporting
System. I don't look for much results from failure
analysis per se except as it contributes to the Failure
Report.

<p style="text-align:center">*　　*　　*</p>

Tests in laboratories do not simulate real applications
and environments. Some sort of technique is required to
define the relationship that exists (assumed) between
laboratory data and the real service behavior.

<p style="text-align:center">*　　*　　*</p>

Development of systematic materials evaluation techniques
should be continued and integrated into structural design
procedures with the goal of developing a highly automated
and interactive vehicle design system.

<p style="text-align:center">*　　*　　*</p>

We feel steps should be taken to implement the NMAB suggestion.
Good accelerated service simulation tests for time-dependent
phenomena such as corrosion, radiation, creep, etc. are
musts. Case histories are good but only half of the
story -- predicting and intercepting new problems is the
other half.

Regarding information centers, a major problem is how to
get people to overcome the old NIH factor. A more modern
version seems to be "I'd rather do it myself".

<p style="text-align:center">*　　*　　*</p>

Data bank idea for materials is a good idea.

<p style="text-align:center">*　　*　　*</p>

5b.　Is "full-scale" demonstration a barrier for new materials?

Yes. The cost of full-scale demonstration is usually so
high that no funds are available for new material tests.
Full scale demonstration tests accomplish little. Usually
they load the vehicle with one spectrum critical for only
one component. Therefore, it has only limited use at a
tremendous cost. Many new material tests of coupons and
components could be made for the same cost.

<p style="text-align:center">*　　*　　*</p>

Yes -- cost.

<p style="text-align:center">*　　*　　*</p>

I prefer to think that a full-scale demonstration is a goal, a successful milestone to be reached rather than a barrier. It also serves as a way to pick up material behavior and processing difficulties that may not otherwise be revealed by small element tests.

<p align="center">*   *   *</p>

Experience with full-scale applications is essential to develop the confidence required for acceptance of a new material.

<p align="center">*   *   *</p>

No! Lack of incentive and opportunities are real barriers.

<p align="center">*   *   *</p>

No, lack of cost data and service experience are.

<p align="center">*   *   *</p>

"Full-scale" demonstration need not be a barrier for new materials. The 5-year delay between the inceptual stage of a new material and its first flying application is largely attributable to the "no one wants to be first" philosophy. R&D support for the user who is willing to be first would encourage more risk taking, since present fixed price contracts penalize risk taking.

<p align="center">*   *   *</p>

5c. How is the materials-structures interface controlled? How can or how should it be?

Usually by poor coordination between some of the necessary parties. It should be controlled by a high level management group with all the necessary parties represented. Because fracture mechanics is so important, the Structural Mechanics representative should be chairman of the group.

<p align="center">*   *   *</p>

Several ways: (1) Material process specifications; (2) Structures Design Philosophy documents at outset of new program; (3) Structures Design Manuals (company) which set forth design allowables, exceptions, qualifying assumptions re. usage of materials; (4) Joint Materials and Structures participation on Material Discrepancy Review Boards and on Structural failure analyses; (5) Joint review of lay-outs and detail design drawings.

<p align="center">*   *   *</p>

Prefer to think of the interface as a dynamic boundary
tending to shift toward structures when well established
materials are used in design -- and towards materials
when new or unusual uses or environments are encountered
with established materials or wholly new systems are used
for the first time. Materials should characterize each
new alloy or material system, particularly regarding
properties other than strength such as: corrosion resistance,
embrittlement susceptibility, weldbondability, fracture
toughness, and protection system requirements.

<div align="center">*   *   *</div>

One way is to have the materials man sign the Engineering
Drawing for approval of materials and processing in
addition to the Stress Engineer's sign-off.

<div align="center">*   *   *</div>

By hard work and recognition of potential problems and
solving them in advance.

<div align="center">*   *   *</div>

Mainly by the structures design group with coordination to
materials and manufacturing.

<div align="center">*   *   *</div>

5d. Are there any subjects related to the preceding topics
which an NMAB Committee could usefully tackle?

Yes. The committee could decide what analysis and what
testing needs to be done to increase the reliability of
new materials for new advanced vehicles.

<div align="center">*   *   *</div>

I'm not really current on what the NMAB Committees are
covering as of today. But -- I'd like to see some
government-industry group address itself more completely
to the following:

(1) The need for and possible ways of implementing
     prototype material applications. (On materials
     research vehicles? More full-scale laboratory
     demonstrations?)

(2) A look at the total aerospace and related industrial
     materials R&D in the U. S. (both government and
     private) with the objective of identifying imbalances.

(3) Consider and propose several realistic approaches to
     risk evaluation and probability of failure and what
     types of data must be generated to make this feasible.

<div align="center">*   *   *</div>

See latest NMAB report on Accelerated Use of New Materials. It has many good ideas that should be followed up.

* * *

6. Regarding more general design problems

6a. Are the concepts of probability of failure and risk evaluation bound to become a routine part of structural analysis?

Yes. Eventually. However, new tools are resisted for long periods of time. For example:

(a) Using Power Spectral Analysis as a recognized tool took about ten years. Even now, certain segments of the industry will not recognize it as a respectable tool, and will not allow specifications to be written which include it as a criteria.

(b) The six degrees of freedom cross coupling analysis was not accepted until a fighter aircraft's vertical tail came off because of the additional cross coupling loads from three to six degrees of freedom.

(c) Flutter analysis, fatigue analysis, sonic fatigue, supersonic panel fatigue, creep, and computer analysis was resisted until they were able to weather the storms of protest of being called "inexact", "costly", "unnecessary burden", etc.

* * *

I would hope so. But this will require an extensive education campaign.

* * *

Yes -- This is what the ASIP, now in revision, will require.

* * *

Yes.

* * *

They've always been. If you mean quantification and documentation in depth, I wonder as to the worth.

* * *

Yes, to a limited extent on components selected for fracture control and only for tradeoff studies of significant factors.

* * *

If they do, we are going to be kidding ourselves and others. I would have no confidence in the numbers. There are just too many variables.

$$* \qquad * \qquad *$$

6b. How can qualitative considerations be quantified for tradeoff and risk evaluation?

The qualitative considerations must be put into a cost effective analysis to determine what quantitative minimum values are necessary. The risk for a new material, must be <u>less</u> for new aircraft. To achieve this lower risk, <u>adequate testing must be done</u> to at least obtain the coefficient of variation of strength at time zero and the residual strength after subjecting a primary structural component to the expected spectrum for the desired service life.

$$* \qquad * \qquad *$$

I doubt if they can, other than by the usual expedient of assigning weighting factors.

$$* \qquad * \qquad *$$

Develop case histories over a period of time, so some feedback is obtained on what is obtained for certain qualitative considerations.

$$* \qquad * \qquad *$$

Depends upon the case. This is called operations research or analysis à la Rand, etc.

$$* \qquad * \qquad *$$

By application studies and tests.

$$* \qquad * \qquad *$$

Waste of time to try.

$$* \qquad * \qquad *$$

6c. What can be learned from the past to make systematic use of available experience?

By doing analysis first, followed by controlled material and small component tests to confirm the analysis, then followed by only two experimental aircraft (one for immediate flight and another for proof testing and detailed instrumental measured flight) new materials with temperature inputs can be understood before larger scale production is initiated.

$$* \qquad * \qquad *$$

First we must document our failures so that we have a "past" to learn from.

<center>* * *</center>

Books could be written on this.

<center>* * *</center>

The $64,000 question.

<center>* * *</center>

6d. What can be done to educate materials engineers in the problems of structural design and structural designers in the problems of materials and processing?

By the services providing institutions and large corporations with funds for this purpose. By making it mandatory for the stress signout of drawings (at some later date -- 5 years from now) to obtain a certificate that they have completed these courses before they can signout any drawings for military aircraft.

<center>* * *</center>

The question implies that this is not being done today. In aerospace -- in my own experience, at least -- I think there is a very good appreciation by the structural designers and the materials engineer of their mutual problems. This may not be so when the men first come out of school but then at this point the men have no identity anyway as either structural design engineers or materials engineers. This is an on-the-job "graduate" training.

<center>* * *</center>

Each must learn enough of each others discipline so they can effectively communicate with each other.

<center>* * *</center>

Have them work together on a design team.

<center>* * *</center>

Age old engineering problem. Colleges and universities fall short here. Companies seldom have time to handle basic problem. Individuals must recognize and do job for themselves. National societies could help if they really wanted to.

<center>* * *</center>

More development time and closer teamwork.

<center>* * *</center>

They can work closer allowing each to get more involved in
the others day to day problems. Each can be rotated into
the others area for a period of time. Periodic meetings
or seminars can be held within the company and during
working hours to discuss materials and structures problems.

*   *   *

Coordination between materials engineers and structural
designers should begin in the R&D planning stages. Awareness
of each others thinking from this early point through
development and application eliminates the pitfalls
encountered when each discipline just goes its own way.

*   *   *

6e. What possibilities exist to break out of the ever-increasing
complexities of our situation?

The complexities will continue to increase. The only hope
is to automate as much of the detail as possible and
integrate all the needed technology into a design system.
Large digitial computers make progress in this direction
possible.

*   *   *

1. Less wasteful practices of the military to buy vast
    numbers of aircraft simulateous with development
    testing. Development testing should be done first
    followed by pre-production prototypes.

2. Funding for educating engineers in multi-disciplines.

3. Symposiums similar to that in Monterey to share our
    problems.

*   *   *

The computer by the storage of data and permiting the
technique of interactive graphics will help us at early
design stages to visually and numerically be able to
determine the effect of a material change on the weight,
shape, cost, and projected life of our design.

*   *   *

None. If continued improvement is desired or required, it
will get more and more difficult to achieve an improvement.
More factors become involved as the structural efficiency
is increased and the structural weight is decreased.

*   *   *

I do not agree that the complexities are ever-increasing
nor that we have a problem of "breaking out" from them.
Certainly, it is true that viewed in total there are many
more materials, more environments, more sophistication in
the determination of design conditions and test evaluation.
But when you get down to the one for one relationship of
the individual engineer to the specific job, the situation
hasn't changed so very much from that of ten or twenty
years ago. The basic approach is essentially the same.
I am an optimist about man's ability to cope.

*   *   *

Just call for and finance new designs and solve problems
on an orderly, continuous basis. Otherwise they'll
accumulate to become bigger than we all are together.

*   *   *

6f. Where do we stand with respect to a clear definition of
test requirements for structural airworthiness (component and
full-scale tests for static and dynamic conditions related to
program development) and for aerodynamic performance (prototype)?

I'd say in an excellent if not overly burdensome position.
Prototype per se are not an answer, especially for time-
dependent phenomena.

*   *   *

Commercial vehicles built at Douglas have had long life
without primary structural failure. We have been successful
because we have relied on thorough analysis, development
and component testing during the design stage, with large
component fatigue tests and carefully instrumented flight
testing to confirm the stresses and expected loads. Full
scale ground tests are too costly for the limited gain
expected. We have used five aircraft to develop 1500 hours
of flight to test the structure and aerodynamic performance.
Therefore we can conclude that military aircraft could use
the same techniques and save a considerable amount of
funds which could be used for new material development.

*   *   *

I interpret this question to mean: What are the essential
differences between a structural air-worthiness verification
program for a production aircraft as compared to a prototype
aircraft? This is being rather thoroughly debated in
industry right now. The trend is probably toward reducing
or eliminating static tests to failure as well as fatigue
tests for those prototype aircraft where essential purpose
is aerodynamic and flight research.

*   *   *

ASD-TR-66-57, "Air Force Structural Integrity Program
Requirements," January 1968 gives the most complete
description of test requirements for airworthiness
throughout the life of the vehicle. What is needed is
more correlation between testing and service experience.
Testing often does not simulate service experience,
particularly with regard to environmental exposure. More
work needs to be done to properly simulate environmental
exposure effects in the laboratory so that corrosion
and fatigue problems can be identified before the aircraft
gets in service. Lack of budget usually limits the
amount of service experience correlation that can be done.

<p align="center">*   *   *</p>

Generally clear definition on structure. Systems funding
should never be committed before prototype evaluations
except in national emergencies.

<p align="center">*   *   *</p>

Airframe problems that are a result of improper structural
tests usually can be tied back to an improper definition
of the environment for design as well as test, i.e., buffet
loads, fatigue spectra, airload distribution, etc. These
factors need defining ASAP so that testing of the airframe
can be done ASAP to minimize the impact of any resulting
changes.

<p align="center">*   *   *</p>

7. What other questions are considered pertinent to design
problems?

1. Should load alleviation and mode stabilization be
   further developed to reduce loads? Would the funds
   used for this project reduce failures more than
   spending funds on structure development?

2. Should more funds be used to develop optimization
   procedures with other disciplines?

3. Should items (1) and (2) be developed concurrent with
   further structural development?

<p align="center">*   *   *</p>

a. Should the DOD encourage application of new materials
   offering significant improvement by assuming more of
   the risks?

b. Should the DOD encourage the use of a material like
    composites on a new major weapon system by paying the
    premium for the higher materials cost even when the
    usual cost-benefit analysis does not favor the use of
    the more costly material?

                        *   *   *

See the latest NMAB report on Accelerated Use of New
Materials.

                        *   *   *

SECTION III

SOME BASIC CONSIDERATIONS AND CONCLUSIONS

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SOME BASIC CONSIDERATIONS AND CONCLUSIONS

1. SYNOPSIS OF PRESENT SITUATION

1.1 Introduction

The objective of this last section is to consider some fundamental aspects of design problems and to arrive at practical conclusions.

The present situation regarding structural design of aircraft is quite unique. Two different and rather contradictory sets of facts are coming together and causing considerable concern. On one hand there are recent developments in high-strength alloys, filament composites, fracture mechanics, finite element analysis, automated design, computer graphics, and other fields which give rise to high hopes and expectations for greatly increased structural efficiency. On the other hand, there have been unexpected difficulties lately regarding detail design and materials applicatication. They have occurred in fields which had been considered thoroughly explored and tested -- except for some seemingly minor modification which may have caused major trouble.

The result is a disturbing realization that the gaps in our understanding of traditional materials and conventional design practices are wider than we thought they were. This has dampened our previously so confident assurance in tackling new developments and has made us wary of extrapolating past experience. In spite of an impressive mastery of sophisticated techniques we are developing a humble readiness to occupy ourselves with some very fundamental considerations about these unexpected difficulties.

The combination of future promises and present problems characterizes the situation in structural design. Promising developments of the future depend on finding solutions for recent difficulties. The difficulties have been caused basically by the increasing complexity of technological developments and it is this rather general aspect which assumes specific importance.

The presentations of Section I give ample evidence for the pervasiveness of design problems. The unsolved questions and various comments of Section II indicate their range. The basic considerations of this Section III will draw attention to the fundamental character of design problems and to the need for a systematic and practical approach toward their solution.

1.2 Survey of Other Findings Concerned with Present Situation

During the past two years several highly qualified committees addressed themselves to specific aspects of recent problems in aircraft structures and in materials application. References 1, 2, and 3 are closely connected with design problems and it is fortunate indeed that they became available just during the final drafting of the present report.

If it would not be for these references, many of their conclusions would have had to be deduced in the present report. Since there is good agreement on all essential points, the established conclusions of these references may serve as the premises for this report which has been somewhat condensed accordingly. A brief summary of these references is given below.

Reference 1 summarizes the lessons learned from structural problems in connection with the F-111 development. It states and discusses the conclusions of the SAB Ad Hoc Committee on the F-111. The following points are particularly pertinent with respect to design problems:

- · Application of fracture mechanics as a design tool holds great promise. Its present limitations must be recognized and a handbook with relevant data and analytical techniques should be published and periodically updated. Formal "Fracture Control Plans" are recommended as part of development programs.

- · A damage-tolerant structural concept is considered to be an objective of vital importance. Periodic structural configuration audits should be accomplished during the development program for all primary structure to identify all aspects of damage tolerance.

- · To exploit the promise of advanced materials while minimizing the application risks, research is required to formulate effective data collection, storage, extraction and presentation methods. A management procedure should be instituted to ensure that materials selection, processing and manufacturing -- the translation from engineering to production -- is under a strict control and audit schedule.

- · Quality assurance and nondestructive inspection must be judiciously defined and rigidly applied. Inspectability and human engineering factors must be adequately considered.

- · Proof-test inspection should not be considered a desirable replacement for quality control and nondestructive inspection.

- Fatigue testing and analysis are the accepted approach for substantiation of fatigue strength. Efforts must be directed to the time compression of such tests, and to the understanding of fatigue damage mechanism.

- Accelerated service testing of two or more aircraft from an early production lot is recommended.

- Technology demonstrator programs, such as the Advanced Metallic Structure ADP, can strengthen confidence in emerging technologies.

- Probabilistic statements about risk can contribute to sounder command decisions regarding development and utilization of weapon or logistic systems.

- Among the critical problems repeatedly encountered during aircraft development and operation, the following are listed:

    misapplication of structural material;
    improper material purchase specifications;
    improper manufacturing processes;
    improper detail design with respect to fatigue;
    poor quality control and reliability;
    inadequate auditing of subcontractors;
    improper analyses and assumptions;
    deficiencies in control and stability;
    change in aircraft usage;
    unanticipated life extensions;
    inappropriate and/or untimely fatigue, static and flight testing.

- Regarding organization and human factors, the need for a high degree of realism during the procurement process is emphasized and more extensive use of independent advisors and advisory panels, with real freedom to speak their beliefs, is suggested.

Reference 2 is a report on the Structural Integrity of Current and Future Air Force Systems. It is based on an extensive team study and the resulting recommendations are directed toward most effective planning, execution, and follow-up of the Aircraft Structures Integrity Program (ASIP) within the procurement agency. Special emphasis is given to the need for making realistic estimates, for understanding the impact of trade-off decisions, and for bridging communication and apparent technology gaps. The benefit which can be derived from independent expert teams and review groups is also stressed.

Reference 3 is concerned with the gap which has developed between the development of new materials and their application in the design of aircraft. It summarizes the findings of the National Materials Advisory Board's Committee on Accelerated Utilization of New Materials. The committee has investigated the causes for the existing delay in the application of promising new materials and concluded that it should be possible to prevent or minimize delays. Its principal recommendation is to establish a continuing function under the auspices of an interagency government organization

- to review the status of new materials and processes;

- to identify those with a potential for wide applicability which can benefit by coordinated support;

- to organize a cooperative program to assure timely application of selected materials.

These three references serve the purpose to evaluate recent difficulties from three clearly defined viewpoints, namely: experience with the F-111 program, need for early identification of structural problems in Air Force systems, and experience with the introduction of new materials. Their recommendations are of special significance in view of the knowledge and experience brought together in the committees responsible for these reports.

1.3 Purpose of Present Project

The present report attempts to take a step beyond the reports summarized in the preceding Section 1.2. As this report forms the final part of the project Interface of Materials and Structures on Airframes, it is based on a more general approach which began with Basic Design Considerations (Ref. 4) and continued with a particular concern about the Decision Process in Structural Design (Ref. 5). These general considerations will now be merged with practical aspects.

The present situation should be quite propitious for combining fundamental and practical aspects. There is a growing awareness that many of our present problems can be solved by improved coordination between the fields of materials, structural mechanics, and design. Such coordination represents a natural process which has been developing slowly over a long time. Much can be accomplished, however, by clarifying and accelerating this process.

The purpose of this report is to evaluate recent design experience from an overall viewpoint and to arrive at practical conclusions. Such an overall viewpoint will coincide with the

viewpoint of the designer who is responsible for the design. It
will combine the three separate viewpoints of References 1 to 3
and will take into account that a practical solution for present
design problems has to be found first, but also that a basic
approach toward the solution of future design problems has to be
established before these problems have grown beyond bounds.

1.4 Method of Approach

Reference 5 recommends further consideration of the
structural design process on an industry-wide basis. In a
modified form and from a somewhat different perspective this was
accomplished at the Monterey Symposium on Design Problems in
Aircraft Structures. With its representative participation
from industry, research, and government agencies, the symposium
provided a balanced perspective for practical and theoretical
aspects.

The proceedings of the symposium, as given in Section I of
this report, serve to substantiate the essential aspects of
design problems. The talks were arranged so that considerations
regarding high-strength steels, fatigue, fracture mechanics,
fail-safe design, procurement policies, space shuttle, probability
of failure, risk evaluation, and technology demonstration were
brought together as basic ingredients of the overall picture.

Resulting questions and comments, as given in Section II,
are obviously only a small part of a very wide spectrum. They
may help to stimulate thoughts and discussions among engineers
and researchers working in these fields. They certainly have
contributed toward putting the results of the symposium into a
wider perspective.

Discussion and conclusions, as given in this present
Section III, are rooted both in the fundamental, but somewhat
generalized, considerations of References 4 and 5 and in the
practical aspects expressed during the symposium. The practical
considerations developed in articles 3.5 to 3.7 contain the
essence of the discussion.

As stated in the foreword, there is no consensus of opinions
in this field and the conclusions must not be construed as
representing the attitude of the Navy Department.

2. OUTLINE OF DESIGN PROBLEMS

2.1 General Remarks

A discussion of design problems should properly begin with establishing the goal of structural design. In its simplest form this can be stated in two words: Optimum structure. The word optimum causes, of course, considerable tribulations. Let us just keep in mind that it comprises all the compromises which are necessary to satisfy the specified strength and stiffness requirements as well as considerations of performance, weight, cost, risk, time schedule, growth potential, maintainability, repairability, inspectability, etc. After some more detailed reflection it will be possible in Section 2.6 to arrive at a more specific interpretation of design goals.

Achievement of an optimum structure is still in the distant future. Optimization procedures loom at the horizon as a major long-range problem and considerable effort is exerted in this field. This long-range problem is inseparably connected with the short-range problem of avoiding the type of technological difficulties which have occurred in the recent past.

Technological difficulties have existed as long as aircraft have been built and they have been overcome reasonably well by different methods at different times. One aspect, however, is new. Our technology has reached a state of complexity where every decision has far-reaching implications. Traditional methods become inadequate when a detail design decision can have tremendous financial consequences, quite irrespective of safety. The cost for rework of a problem of fatigue or stress corrosion encountered during the guarantee period of an aircraft may exceed the financial resources of a company.

2.2 Basic Concepts

For the following discussion it may be helpful to begin with establishing two basic concepts which will be used in connection with structural design.

a. Overall Responsibility of Structural Design

Structural design has many facets. It includes advanced design which is concerned with establishing design concepts and selecting materials and structural configuration. It also includes detail design where the decisions made in advanced design are translated into final working details and into full substantiation of airworthiness and other considerations.

The full concept of structural design must refer to everything connected with the load-transmitting structure. It includes selection and optimization of material, structural configuration, and design details; substantiation of airworthiness by analysis and tests; and determination of weight, cost, reliability, fabricability, inspectability, and maintainability.

This indicates that structural design has to bear full responsibility for all the complexities and consequences of technology. Anticipation of cost overruns as well as difficulties in scheduling, materials processing, fabrication, inspection, and maintenance must be included in this responsibility. Much of it has previously been left to manufacturing without giving it proper representation during the decision-making process.

This overall responsibility of structural design is the clear lesson learned from the experience of recent years. It is emphasized here as a basic concept because the full significance of this lesson is not yet completely appreciated within the engineering community at large.

b. Team Work within Structural Design

The second aspect is a consequence of this first consideration. It is generally not possible to combine all the knowledge, experience, and skill required for a major design component in a single individual. A team effort is required and responsibilities must be clearly delegated. There are three basic fields which are closely related but distinctly separated by educational background:

Materials engineering will be responsible for materials properties and characteristics, processing, fabricability, inspectability, and maintainability as well as materials testing, evaluation, application, maintenance, and follow-up procedures, i.e. all aspects of materials behavior from cradle to grave;

Structural mechanics will be responsible for static and dynamic analysis with respect to strength and stiffness as well as for testing of structural components, i.e. all aspects of airworthiness at minimum structural weight;

Design will be responsible for the traditional field of detail design as well as all the considerations of overall concepts and trade-offs regarding cost, risk, time schedule, and the various "-ilities", i.e. all aspects of coordination and optimization.

These three disciplines are integral parts of structural design. There will always be some overlap along the border-lines of these interrelated fields but we have to realize that structural design is an entity and requires a team effort which is basically different from adversary confrontation of various disciplines. Overall responsibility for guiding this team effort must be clearly assigned but, beyond this, each member of the team must be aware of his responsibility as an integral part of the total effort. The need for such a spirit of common responsibility should be recognized as another basic concept.

2.3 Design Problems in the Realm of Technology

Having established the basic responsibility of the design team for every aspect of structural design, it becomes apparent that we cannot be satisfied with considering technological problems only. They may serve as a starting point and it will be practical to consider basic design problems as falling into three groups: those within the realm of technology, those beyond the realm of pure technology, and those which are problems of policy but have a direct effect upon design.

Problems of a technical nature may be categorized as follows:

a. Basic Mechanics of Failure

There is still a fundamental lack of scientific knowledge and understanding regarding mechanics of failure. This is particularly noticeable in the fields of fracture mechanics, fatigue, and stress corrosion. The necessary research must take place in the field of materials science and is, although of basic importance to structural design, beyond the jurisdiction of the design engineer.

b. Materials Processing, Manufacturing, and Inspection Methods

Limited knowledge about fundamental aspects of failure has been a frequent cause for trespassing unwittingly into forbidden zones during processing and manufacturing operations. Many typical examples are discussed in Section I and many more could easily be found. Quenching rate, residual stresses, hole preparation, change of vendor, nondestructive testing methods are just some of the potential pitfalls.

Extensive testing with a great number of parameters is usually required. The large test programs on basic fracture mechanics data and on spectrum/environmental effects mentioned in Section I in the presentation by W. C. Dietz for just one aspect of one material indicate the magnitude of the task. The

test data on K_{1c} values shown in the presentation by W. S. Hyler indicate how much scatter in test results can occur within given specifications and how difficult it may become to interpret test data correctly.

From the perspective of the materials specialist, clearly formulated questions about the characteristics of new materials can be answered by systematic testing. However, when the materials engineer becomes a member of the design team, his problem is to recognize all potential modes of failure and to anticipate any difficulties which may develop in manufacturing and quality control. A multitude of environmental and operational conditions, varying from one component to another, result in many combinations of temperature, exposure time, stress, sequence of cycling, corrosive conditions, etc. Slightly modified processing techniques may influence test results greatly. All this means that definition of significant test conditions as well as evaluation techniques assume major importance.

The sheer magnitude of required materials data for application of a new material is immense. The corresponding problems are well known to the materials community and further detail discussion would go beyond the scope of this report.

c. Application of Recent Technology and Techniques

Newly developed high-strength materials and corresponding manufacturing technologies pose innumerable problems with hard-to-predict consequences regarding crack initiation, hydrogen embrittlement, stress corrosion, etc. Typical problems in this field are discussed in several presentations of Section I.

W. H. Sparrow shows illustrative examples for high-strength steel parts which failed unexpectedly. J. C. Ekvall starts with typical problems in fatigue and surveys the present state of the analytical airt, after a decade and a half of intensified development in this field. W. C. Dietz presents typical problems in fracture mechanics and gives an outline of present techniques in this field which has come into its own only very recently.

W. S. Hyler shows another problem which is a typical example for the need of full coordination between new technologies in materials engineering, structural mechanics, and design. Slightest variations in materials processing can cause large scatter of K_{1c} values, affecting structural analysis and fundamental aspects of fail-safe design.

A large part of the present research effort is directed toward further development of fracture mechanics as an important analytical tool and toward far-reaching application of damage tolerance as a basic concept in structural design. From a wider perspective this appears as the latest, but certainly not the last, of a long line of design problems which have included stress corrosion, fatigue, integral structures, thin-sheet design, etc.

d. Future Technology

Technological problems of a new type will have to be faced in connection with high-temperature applications. The presentation by F. F. W. Krohn is concerned with the new tasks which may confront us for a space shuttle in the fields of materials, structural mechanics, and design. Full evaluation of the experience gained with recently developed technology and a methodical approach will be a prerequisite.

There is also the field of filament composites. This has not been included in the presentations at the symposium to avoid distraction from basic issues. Much specialized development work is required but systematic progress is made since the importance of this field has been generally recognized.

Another field which should be mentioned in connection with future technology is structural optimization. This aims at the very core of the designer's problem, namely how to obtain an optimum structure. Considerable difficulties exist in the fields of mathematical programing and search methods and work proceeds along various lines.

e. Communication Within a Discipline

Technological information to keep abreast of newest developments is not easily accessible, particularly to the engineer fully involved in everyday work. Frequently there is a flood of information but research results are published in many different places, practical experience of others becomes known belatedly and in rather incoherent form, and clear conclusions are disseminated only slowly. Competitive considerations can also have a retarding influence.

Although interest in new fields grows rapidly -- e.g. fatigue one-and-a-half decades ago, stress corrosion in the early 1960's, or fracture mechanics now -- it is a slow and uneven process to arrive at accepted standards. Again this is pointed out quite clearly in several presentations of Section I.

W. H. Sparrow illustrates how experience with high-strength steels had to be accumulated the hard way. J. C. Ekvall shows that determination of fatigue life is not yet an exact science, in spite of a tremendous effort spent on it, and depends on experience with design details on previous structures and with previously established stress levels. W. C. Dietz demonstrates in the field of fracture mechanics how practical experience is related to the development of new methods. W. S. Hyler indicates the role of experience in materials processing. Everywhere costly mistakes could be avoided if systematic information about previous experience were available.

The problem of communication within a technical discipline can be considered to be a technological problem. However, it becomes apparent that it cannot be solved on the level of the engineering specialist. This points toward further problems which are beyond the realm of pure technology.

2.4 Design Problems Beyond the Realm of Pure Technology

The preceding problems in the realm of technology can be recognized easily. There are other problems, however, which are not so clearly visible and which are in an ill-defined region beyond pure technology. They may be considered in the following problem areas:

a. Communication Between Technical Disciplines

The difficulties of communicating and keeping informed within one's own technical discipline were discussed under 2.3e. Design, however, involves several different disciplines and communication between them can become formidably difficult. Materials engineering, structural mechanics, and design are closely related but the difficulty of communication between them is emphasized strongly in several of the presentations in Section I.

W. H. Sparrow concludes his presentations about high-strength steels with the warning that there is no substitute for communication between engineering, manufacturing, and quality control. W. C. Dietz shows throughout his considerations about fracture mechanics how important it is to have a full exchange of specific information between engineering, manufacturing, and inspection. W. S. Hyler begins his comments on fail-safe design with pointing out the need for interrelation between designer and materials and process engineers for any consideration of fracture and fatigue crack propagation.

Considering a large number of typical design problems, it appears that a lack of inter-disciplinary communication can be found either as their cause or, at least, as a contributing factor. Such a statement should not be taken lightly. It indicates that no amount of specialization can solve present design problems unless the specialist develops an understanding for interaction between his field and others.

There is nothing new about the need for communication. It has existed as long as there has been specialization. Some progress has certainly taken place during the last decade but this has not overcome the basic fact that different specialists have different concerns and are not readily aware of each other's line of thinking. Team work in structural design requires full communication and understanding of interaction to a degree which has not yet been developed.

b. Risk Evaluation

New developments in aircraft structures involve some risk. There is, first of all, the technological uncertainty which may be expressed as probability of failure. Beyond this, there is the risk of exceeding cost estimates and time schedules. All these aspects have to be incorporated in risk evaluation. Some of the inherent difficulties are discussed in Section I.

J. E. Fischler substantiates an approach to use probability of structural failure for the comparison of different designs. Developments in fracture mechanics make it possible to determine the probability of structural failure as a function of several design parameters. This concept can be used to compare materials and types of construction on an equal basis.

W. E. Ellis indicates some tentative steps toward overall risk evaluation. Much can be learned from procedures which have been developed by operations analysis. It will be a long way to transform this into a useful tool for structural design. A first and very important step is to make the designer aware of the line of systematic thinking which has been developed in this field and which can supplement his engineering techniques in a significant way. Beyond this, much thought will have to be given to the expression of qualitative considerations in quantitative terms.

c. Ideas and Decisions

Each of the problems in structural design provokes creative thinking and, because several solutions are generally possible, calls for decisions in the face of uncertainties. This requires a systematic approach in addition to technological expertise. Structural design has to be rooted in both technology and methodology.

The decision process in structural design has been discussed in Ref. 5. It will have to be based on analytical models which combine the considerations of airworthiness and optimization. Models as shown in Fig. B-1, C-23, and C-24 are typical steps in this direction. Development of such a decision process is the goal of design methodology. It has to take place in step with the solution of the other basic problems in structural design. A fundamental need is to provide visibility and clarity for any design decision.

2.5 Policy Problems Affecting Design

The design problems discussed on the preceding pages give full regard to the engineering viewpoint even if some of them cannot be solved on an engineering level. There are other decisions, however, which may have a far-reaching influence on structural design without giving full cognizance to the engineering viewpoint. These decisions are usually made on a management level where engineering considerations are balanced against various aspects of funding, timing, and general policy. The two subjects of procurement policies and test programs deserve special attention.

a. Procurement Policies

Procurement policies can have a pronounced effect on structural design and the following discussion may help to clarify some of these aspects.

Throughout the 1960's there was a trend toward increasingly rigid contracts at fixed price, with a total package incorporating R&D, production, performance, and time schedule. It was only around 1970 when some fallacious reasoning in this trend became apparent. Procurement agencies were driving toward unrealistic requirements to obtain maximum performance while bidders made over-optimistic estimates in a highly competitive environment. The risks of an evolving technology were not appreciated properly and resulted in huge R&D and production costs to solve unforeseen difficulties. Structural design conservatism was frequently squeezed thin between technological need and available budget or had its flexibility for trade-offs curtailed by detail specifications.

Important lessons have come from this type of experience and may be summarized as follows:

- Pre-contractual assessments require a high degree of technological and budgetary realism and objectivity both on the side of procurement agency and prospective contractor.

- Weight, cost, and time schedule require closest control but should not be considered as fixed quantities by themselves. They are means to an end and trade-offs must be encouraged in order to obtain an optimum total design rather than a design which is best in one aspect at the expense of another.

- Costs for research and development must be separated from production costs and only the latter should be subjected to penalties and incentives.

- Pre-production concept proof should be emphasized but it must be realized that this is not as simple as just having a prototype. Time dependent phenomena or production feasibility will not necessarily be demonstrated in prototype programs.

- In case there is concurrent development and production, the inherent risks must be clearly recognized and accepted.

Due to this interrelation between structural design and procurement policies, future policy developments will have to be scrutinized closely. The trend seems to be toward providing a cost-reimbursement basis for the R&D phase with its higher risk and a fixed-price-incentive basis for the production phase. Much emphasis will be on milestone demonstrations as well as cost and schedule control.

The future will probably also hold a shrinking market with fewer but more sophisticated projects. There might be a recourse to prototypes by several companies with the production contract awarded after technical demonstration.

From the viewpoint of structural design there are two particularly important aspects to be emphasized in connection with any procurement policy:

- The risk of new technological developments must be clearly recognized and potential difficulties should not come as surprises;

- The program for structural testing has to be prepared carefully not only with respect to its scope but also with respect to its timing.

Risk evaluation and a reliable estimate of structural testing requirements are important aspects of a procurement policy.

b. Structural Testing

Structural testing is, of course, a straightforward
engineering function. However, it is listed as a policy problem
because the necessary funding depends on policy decisions.
There will always have to be a compromise between the engineer's
desire for verification by testing and the manager's reluctance
to provide the considerable funding. Merely to set up a
consistent program for structural tests has become a task which
is not easily done. There is also the additional aspect that
expenditures for testing can result in significant economic
advantages in production as well as in improved analytical
techniques.

Structural testing is an integral part of structural
design. With increasing emphasis on environmental conditions
and on damage tolerance the amount of testing can grow excessively.
Yet new technology requires extensive and systematic testing in
order to reduce risk to an acceptable level. Agreement between
analysis and experiment is the basis for our confidence in the
integrity of a structure. Much engineering work is still
required to obtain closer coordination between analysis and
testing and to save time, money, and talent as analysis may
eliminate some testing.

Component testing is caught in the dilemma that it can
start only after the component has been designed and manufactured
but that tests should be finished in time so that any modifications
do not affect detrimentally the production process. In view of
the advanced planning necessary for production, the necessary
compromises have to be made within the context of overall policy.

Within this need for compromising, the structures
engineer has the responsibility to recognize clearly what kind
of information is required from what type of testing in order to
substantiate airworthiness for all operational and environmental
conditions. In this connection it is important to evaluate
critically any prototype testing to see whether it plays an
important role within the structural testing program.

These few remarks may suffice to show the close
relationship which has to be established between structures
engineering and management, starting at the very beginning of a
project. Recent experience has shown that funding is always
available if a panic situation should develop but that a
comparatively small amount of expenditure may prevent such a
situation.

2.6 Designer's Viewpoint

The preceding broad-brush treatment of problem areas in
structural design requires, of course, much amplification before
it can be thoroughly interpreted. However, in spite of its
briefness, it seems to be adequate for the purpose of clarifying
and illustrating two basic aspects: Design problems are
rooted in technological difficulties but they branch out into
wider fields.

After having considered separately those problems which are
of a specialized technological nature and those which go beyond
pure technology, it can be recognized that each of these two
categories has several facets. Instead of expressing the two
categories as a field of technology versus another field which
is beyond pure technology, we may think in terms of airworthiness
versus optimization, or specialization versus interrelation.
Each of these terms implies a different aspect, and indicates
the many-sidedness of each type of problems.

Technological problems can be seen clearly after they have
developed -- even if it may take a post mortem in extreme cases.
These problems have a direct effect on airworthiness, and steps
toward their solution are taken quickly. Responsibilities
are distributed among well-defined disciplines or organizational
groups and steady progress toward the solution of these problems
can be expected. There seems to be no need to pursue them any
further in this report.

The picture is different, however, for problems reaching
beyond the realm of pure technology. These problems are
ill-defined and at the same time most pervasive and elusive.
Although they are of a very different kind than the well-defined
technological problems, they may easily permeate any of them.
It became quite apparent from the presentations by W. H. Sparrow,
W. C. Dietz, and W. S. Hyler how the work of the specialist
has to be correlated with inter-disciplinary communication,
and from the presentations by J. E. Fischler and J. W. Ellis how
probability of failure and risk evaluation may play an important
role in the decision making process. This type of problems
reaching beyond traditional aspects of pure technology forms
the very essence of design because design cannot be satisfied
with just finding a technical solution. It has to strive for
an optimum solution considering all circumstances.

Much general consideration has recently been given to
design objectives and methods. Several books on design
methodology have been published just during the past few
years. Combining quotations from various authors, design must
not be confused with art or science or a form of mathematics,

but it is a hybrid activity which depends on a proper blending of all three. We may think of it as a creative, goal-directed, problem-solving activity which depends on decision-making in the face of uncertainties and is concerned with all aspects of a problem.

Although design is a creative activity, it has to submit itself to a rigorous logic. A large number of ideas have to be analyzed and evaluated. Both the climate for encouraging new ideas and the decision process for evaluating the implications of these new ideas have assumed much importance for the solution of design problems.

This kind of considerations leads toward a more specific interpretation for the goal of structural design. The dual nature of design problems -- specialized technology versus optimization and complex interrelations -- is of basic significance. We have to realize that most of our present design problems can no longer be solved by merely concentrating our efforts on specialized technology. Full consideration must also be given to the interrelation between specialized fields -- both for optimization and for assurance that no aspect of airworthiness has been overlooked.

The following part considers these not so obvious yet completely vital aspects of structural design problems.

3. UNEXPLORED ASPECTS OF DESIGN PROBLEMS

3.1 General Remarks

The preceding consideration of design problems from the designer's viewpoint drew attention to the need for going beyond the traditional concern about specialized technology. Four questions have particular significance:

- How can communication between various specialists be improved?

- How can risk be evaluated?

- How can the decision-making process be clarified?

- How can available information be made more accessible?

Answers to these questions cannot be found on a purely technological level. They require full consideration of educational and organizational aspects.

The recurrent theme contained in these questions is the need for providing full communication and mutual understanding among all members of the design team. This would sound like a self-evident and rather superfluous statement if it were not for the overwhelming evidence -- expressed in many presentations of Section I -- which indicates how hard it is to accomplish this communication and mutual understanding and how close this point comes to the roots of many or even most of our design problems.

The implications of this statement must be recognized. The problem cannot be solved on paper. The objective will be accomplished only when attitudes and actions of engineers express that they think not as engineering specialists but in terms of the overall design project. Our conventional engineering education has not prepared us to do this.

The following considerations begin with some basic aspects of engineering curricula, continuing engineering education, and engineering professionalism and lead up to a practical approach regarding some very fundamental features of design problems.

3.2 Engineering Curricula

Engineering is based on science and the scientific approach consists of analytical and experimental techniques applied to a clearly defined problem. Education along these scientific methods has been the foundation for outstanding technological achievements

but unfortunately also for a lack of success in translating specific achievements into an overall entity. We have to learn how to coordinate diverse and frequently contradictory requirements from many specialized subjects in order to obtain an optimum overall design.

Contrary to the typical problems in science, a typical design problem is not clearly defined. It is the responsibility of the designer to recognize all significant parameters and to define the problem before the scientific process of analyzing it can take place. Usually there are several possible solutions and the designer is responsible for determining which is the optimum among them. Our engineering education in the past two decades has done exceedingly well in preparing the student for the analytical process of problem solution but has usually neglected the basic design aspects of problem definition and optimization.

A first reaction to the long trend of putting so much emphasis on scientific specialization became visible in the mid-1960's. Development of inter-disciplinary graduate courses was pioneered at MIT and Stanford and sponsored further by NASA. These courses contributed much to a growing awareness for the need of coordination between academic disciplines.

However, these courses were mostly oriented toward advanced design concepts and were not particularly concerned with the interaction of materials and structures, i.e. structural design. The AIAA Aircraft Design Committee, among others, has been concerned about the vanishing of design from aero curricula (Ref. 7). More than a quarter of the aero curricula seem to have eliminated design courses entirely and only less than a quarter devote a minimum of four semester hours to design. The need for much closer cooperation between aerospace industry and academic community is an obvious conclusion.

The present situation is full of contradictions. In spite of the gloomy outlook of Ref. 7, there are a good number of very promising starting points. Most of them are outside the field of aeronautics. The Engineering Development Program of the University of California at Los Angeles and the methods of Case Studies as developed at Stanford and Berkeley may be mentioned.

There seem to be three basic difficulties in making our engineering curricula responsive to the needs of industry. Firstly, any changes in academic life take place at a slow pace. Secondly, no clear guidelines have been established in spite of much interest in the subject. Thirdly, particularly in aeronautics, engineers in industry and faculty members at universities who may have recognized the problem have been absorbed so much in their specialized fields that they did not find the necessary time to do something about this overall aspect.

3.3 Continuing Engineering Education

Continuing education is based on the recognition that technological developments make an engineering education obsolescent after not too many years. Evening classes, company-sponsored courses during working time, lectures and meetings, and full-time short courses are most frequently used to keep the engineer abreast of technological developments.

All this is being done in many modifications, depending very much on special conditions. The spectrum ranges from courses which are given to obtain an academic degree to others which are directed purely toward professional development. Educational media are increasingly employed.

For instance, in regions where several aerospace companies and a university are located, lectures given on the campus are brought by TV into classrooms inside the companies during working hours, frequently with two-way communication between instructor and each individual. Such courses are usually part of an academic curriculum and they are taken for credits as the participants are working toward an advanced degree.

On the other hand, full-time courses which may extend over a few days or a few weeks usually are directed toward a subject of specific professional interest of an advanced nature. Such courses are given by a group of specialists from a viewpoint of sharing information on recent developments and the participants are experienced engineers. The purpose is clearly continuing education in the basic sense of the concept.

It can easily be seen that there is much more flexibility in this type of continuing education than in courses which are part of a formal curriculum. Academic credits lose their significance, courses can be tailored much more readily to the needs of industry or special developments, and instructors may be chosen in accordance with their specific competence, whether they come from faculties or industry or research.

Much is going on in this field of continuing education. Contrary to the slow changes in well-established curricula of formal engineering education, everything in continuing education is in a fluid state of early development. There seems to be a unique opportunity to apply some of these developments in the educational field to the design problems in aircraft structures.

3.4 Engineering Professionalism

Engineering graduates going into industry have generally a clear analytical mind and are well equipped to solve problems in their fields of specialization. Some of these young engineers are outstandingly bright but the process of advancing into

engineering positions with broad responsibilities is slow and
tedious. Usually it takes the route of proving their excellence
in a specialized field and even for capable engineers it may take
a good deal more than 10 years before they begin to develop an
understanding for the all-important interrelation between
various fields of specialization. Basically they are left to
their own devices how they go about it.

There are, of course, highly competent engineers in industry
with many years of experience who have grown beyond their field
of specialization and have learned to communicate with adjacent
disciplines. They are in responsible positions and generally
overburdened with work. However, for each one of them there are
a great number of others in various stages of the laborious
process of trying to accumulate some experience beyond their
own field in order to broaden their horizon and make them more
valuable engineers. Would it not be plain common sense to help
accelerate this important process?

Let us look just at the typical problem of communication
between structures and materials engineers. When the structures
engineer obtains a K_{1c} value from the materials engineer, does he
understand sampling techniques and processing tolerances on which
the value is based? When the materials engineer proposes the use
of a certain heat treat, does he understand all implications with
respect to residual stresses, environmental and operational
conditions which may occur? The specialist's jargon frequently
expresses concepts which are not readily explained. Any attempts
of explanation may leave some essential detail misunderstood due
to differences in viewpoints and lines of thinking. Section I
contains many examples for design problems which were caused by
this type of difficulties in communication between specialists.

Management is, of course, vitally interested in improving
this communication process. So is the individual engineer
because it enhances his professional growth. No engineer worth
his keep wants to be just a cog in a big machine. A pure
specialist might too easily be unemployable as soon as he becomes
unemployed.

3.5 Practical Considerations about Educational Aspects

We have seen that our background has preconditioned us to
think as specialists rather than in terms of an overall design
project. On the other hand there is a basic need and a willing
readiness to recognize the role of complex interactions and to
develop new methods to deal with them. The question is how this
can be accomplished.

An answer can be found along the lines of continuing education. There is, however, one important qualification. Contrary to other fields, the subject matter for this type of course cannot be prepared on the usual academic level of research and special-ization. It rather has to be based on a systematic evaluation of recent practical experience in structural design.

This experience has been gathered in industry. Therefore, the subject has to be evaluated and written up by engineers who are thoroughly familiar with recent design experience and who understand the far-reaching implications. Nothing of this sort has been done because it requires a joint effort of considerable magnitude to prepare the necessary outline, subject matter, and text material for such a course.

If the development of a course program and the corresponding text material is sponsored by a government contract, it can be done thoroughly and will be available to the entire industry. The project is too important and too urgent to be left to a somewhat haphazard development on local levels. On the other hand, when a well-prepared text is available, qualified instructors can be found locally. The text will serve as the backbone for courses or seminars in continuing engineering education given throughout industry or for self-study. A loose-leaf textbook -- to be kept up to date -- may be the most practical format, but some other methods of educational media may deserve consideration.

Such a project requires the cooperation of several highly competent and motivated engineers. It will have to start with systematically extracting, describing, documenting, and evaluating recent experience. This has to be translated into a form which can serve to prevent repetition of past mistakes, to provide an introduction into newly developing fields, and to direct attention toward new methods.

It appears that a course subject of prime importance will be Interaction of Materials and Structures. This requires the cooperation of engineers experienced in structural design, analytical methods, materials characteristics, processing methods, manufacturing, and inspection. Improved communication and interrelation between specialists is based on a basic understanding of underlying principles, applicable methods, significant aspects, and recent developments in adjacent fields. This means familiarity with each other's outlook and methods of approach. Each specialist has to put himself into the shoes of other specialists and has to explain to them some basic aspects of his own field. Emphasis will be on those aspects which have contributed to recent design problems and which have to be understood by other members of the design team to prevent interface problems. The text can be tailored to the practical needs of a design team. It is this type of information which is not available in a systematic form anywhere and which could prevent a large percentage of our typical design problems.

Other course subjects should be Risk Evaluation and Decision Making. These are strange fields to most engineers and communication is correspondingly aggravated. In the past these subjects have not played any role in design -- except in some of the more abstract aspects of parametric performance studies and fatigue. With increasing complexities a close interaction between engineering and operations analysis, extending to the level of structural design, cannot be avoided much longer. The engineer needs an introduction into operations analysis from his design viewpoint to familiarize himself with basic possibilities, methods of approach, and practical applications so that he does not violate elementary rules and can recognize when there is need for specialist advice. New possibilities can be explored only if the engineer is able to communicate with the operations analyst.

Such educational efforts within the framework of continuing engineering education are aimed at engineers in industry. This should produce early results. For thorough results, however, the aeronautical curricula at universities have to be affected and closer coordination between universities and industry is necessary. It seems that an advisory group made up of members from universities and from industry could exert a very healthy influence. Such a group would have to provide guidelines firstly to bridge the gap which has developed between engineering needs of industry and scientific orientation of engineering curricula, and secondly to coordinate the efforts which will be required in the field of continuing education. Many implications are involved and the full spectrum of education for aerospace engineers must be taken into account. University curricula and continuing education have to be coordinated as two fields of fundamental importance.

3.6 Practical Considerations about Information Systems

After having considered educational aspects as a prerequisite for dealing with complex interactions, attention must also be paid to another side of the problem. Much waste and frustration occur when basically available information is not accessible for secondary reasons -- which is a problem of organization.

This has been a particularly blatant problem in the field of materials characteristics where test data are produced in many places but become meaningful only when correlated with other data and evaluated with respect to clearly specified test conditions. The enormous quantity of data being developed makes it imperative to have clearly assigned responsibilities for collecting, interpreting, storing, and disseminating this information.

The Defense Metals Information Center of Battelle represents a basic step in this direction. It incorporates the essential capability to interpret and evaluate data. Yet much additional effort and funding are required. Decentralization in accordance with available talent and facilities is quite feasible. Reference 4 discusses some aspects of a materials information system and computerized methods, and Ref. 1 also points out the need in this field. The dominant need for such centers of information as a prerequisite for a healthy aerospace industry is easily apparent and it can only be hoped that the various obstacles will be overcome in the near future.

Reference 2 recommends an analogous step by establishing a Structures Information and Analysis Center "to collect, process, investigate, analyze, evaluate, disseminate and advise on structural materials applications, analysis methods, test techniques and failure modes and causes".

Another step consists of handbooks containing up-to-date techniques in newly developing fields. Most major aircraft companies have developed manuals of this type in fields like fatigue and stress corrosion. Much duplication of efforts could be avoided and more complete information and consistent application could be assured if such handbooks would be sponsored on an industry-wide basis. Reference 1 recommends particularly such a Handbook on Fracture Mechanics for Aircraft Designers, periodically updated as new data and experience become available.

A further step should consist of Case Studies, describing the full history of significant failures and design problems which have been encountered and solved in industry. At present this is done to a certain extent. Important failures result in engineering reports which have restricted circulation. Some basic aspects eventually are filtered into technical papers or articles. Other aspects enter into a grapevine system. However, the full information should be available to all those engineers who may learn from this experience to avoid similar mistakes. To write up a comprehensive case study is a major task and Reference 4 discusses the practical aspect of having this done by graduate students who can gain much insight into design problems by doing this.

3.7 Practical Considerations about Overall Perspective

The preceding considerations about educational aspects and information systems indicate some stimulating and far-reaching possibilities which are completely within practical reach. Their realization, however, requires an effort which can be exerted only when the importance of design problems is considered from a long-range perspective.

Such a long-range viewpoint has been taken for the USAF program on Advanced Metallic Structures. This program is described by D. A. Shinn in his presentation in Section I. It is based on the recognition that there is no systematic approach available to structural design at increasing complexities and the program is directed toward establishing a practical approach for solving inherent problems. Special emphasis is given to an efficient system for distributing the resulting information to the entire technical community.

This large-scale effort toward finding practical solutions for design problems represents an important step. To be fully effective, however, it must be coordinated with a corresponding effort regarding fundamental considerations. An educational program as outlined under 3.5 can prepare the ground for such fundamental considerations and an information system as outlined under 3.6 can remove basic obstacles in the path of solving design problems.

It appears that an educational program and an information system along these lines should be considered as an important complement to the program on Advanced Metallic Structures. Such an approach toward three esential aspects -- hardware, software, and education -- together with the well-recognized need for technological research and development, would provide a logical and promising course of action. The necessary funding for the data information system will be considerable but can be spread out -- besides, any delay will increase the eventual costs. For all other aspects suggested under 3.5 and 3.6 only modest funding is required.

As an encouraging omen it may be mentioned that the USAF program on Advanced Metallic Structures as well as the report by the Ad Hoc Committee (Ref. 1) and the Study of Aircraft Structural Integrity (Ref. 2) emphasize the need for a systematic exchange of information. This recognizes an attitude that problems of structural design should be beyond competitive considerations. A structural failure in one aircraft hurts all others. The whole aircraft industry is in the same boat and everybody suffers when somebody contributes to a leak.

Technical competitions will be governed by the quality of a design team and the corresponding probability for a successful design. To build such a team, to provide stimulating working conditions, and to instill a creative spirit takes a long time. Technological expertise can be acquired by hiring a few experts. However, an awareness of complex interactions must be developed methodically and still begs to be recognized as a fundamental aspect of design problems.

*   *   *

There is an additional aspect which is worth mentioning as we are looking at design problems from an overall perspective. An important clue is provided in the conclusions of both the Ad Hoc Committee (Ref. 1) and the Study of Aircraft Structural Integrity (Ref. 2) which emphasize very strongly the benefit which can be derived from an independent group of experts.

Independent of these conclusions, the principal recommendation of the Committee on the Accelerated Utilization of New Materials (Ref. 3) consists of having the function of such a group in the field of materials. Correspondingly, practical considerations about educational aspects (see art. 3.5 of this section) point toward an advisory group in the field of engineering education. Some "practical" people may say that this is wishful thinking. Yet in the field of aircraft structural integrity, where considerable obstacles of a competitive nature had to be overcome, an industry-wide group of experts has been in existence for a decade, steadily growing in importance.

Much grief, frustration, and waste could be avoided and many potential design problems of the future could be prevented from developing if advisory groups of independent experts would be available to provide guidance in the three fields of materials, structural integrity, and engineering education. Even if this is only an advisory function, such panels can exert a great influence if they represent the proper blend of experience, realism, and vision.

4. CONCLUSIONS

Design problems in aircraft structures can be considered as belonging to three different categories. Each of them requires efforts of a distinct kind in order to solve present and future difficulties.

a. Technology

On a technological level additional research and development is required regarding

- mechanisms of failure, particularly in the fields of fracture mechanics and fatigue;

- materials processing, manufacturing, and inspection methods;

- application of recent and future technology and techniques.

Necessary efforts in these fields have been generally recognized and identified. Systematic progress will depend on the available funding for research and development programs which have been outlined by cognizant agencies.

b. Technological Organization

Some basic design problems cannot be solved on an engineering level. They require an organizational effort by government agencies and top management.

- Data information systems are necessary to assure that available data become accessible to the engineering community. DMIC of Battelle and the planned Structures Information and Analysis Center of AFFDL are steps in this direction but much additional effort is required (see 3.6).

- Handbooks containing up-to-date techniques in newly developing fields should be sponsored on an industry-wide basis (see 3.6).

- Case Studies investigating all aspects of recent failures and complex problems which have been solved can serve as lessons to be learned by the industry. They should be written up systematically and circulated widely (see 3.6).

- Interaction between management decisions and design problems requires particularly close attention in the fields of procurement and structural testing (see 2.5).

c. Engineering Education

On an educational level the engineer needs a helping hand to grow beyond his field of specialization and to understand the complexities of technological problems.

- Early results can be achieved by sponsoring the development of a course outline and corresponding text material on Problems of Interaction between Materials and Structures.

- General awareness of unfolding new possibilities can be stimulated by additionally sponsoring the development of a course outline and corresponding text material on Risk Evaluation and Decision Making in Engineering.

- Long-range results can be influenced by an advisory group on engineering education, representing both industry and universities.

Details are discussed under 3.5. The funding required for this educational effort is small compared to the large effect it will have on the engineer's approach to complex design problems.

*   *   *

Design problems can be prevented before they develop. This requires an approach where individual engineer, engineering community, management, and government agencies have to pool their competence, resources, and initiative. Increasing complexities may result either in challenging tasks which can be met or in frightful nightmares which are hopelessly entangled. There still seems to be a promising opportunity to influence these developments.

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13. ABSTRACT

The proceedings of the Monterey Symposium on Design Problems in Aircraft Structures provide a basic survey of design problems from the engineer's viewpoint. Further analysis of the present situation draws attention to some essential aspects which are not yet generally recognized. This leads to the conclusion that recent design problems cannot be solved on a technological level alone. An organizational effort is needed to disseminate available information. Beyond this, the complexity of interactions must be understood more thoroughly and this requires an educational effort on a broad basis. A practical and systematic approach toward the solution of these problems is developed.

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